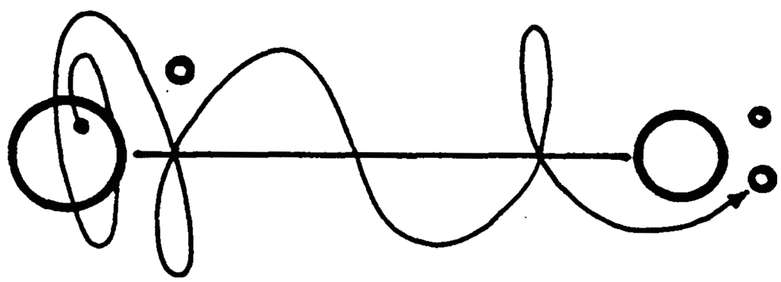
NGT-21-002-080
NGT-80001

MANNED MARS EXPLORER PROJECT



Guidelines for a Manned Mission to the Vicinity of Mars Using Phobos as a Staging Outpost; Schematic Vehicle Designs Considering Chemical and Nuclear Electric Propulsion

16 MAY 1988

SICSA
Sasakawa International
Center for Space Architecture

Room 122 ARC, University of Houston
4800 Calhoun, Houston, Texas 77004

UNIVERSITY OF HOUSTON
COLLEGE OF ARCHITECTURE

(NASA-CR-184745) MANNED MARS EXPLORER PROJECT: GUIDELINES FOR A MANNED MISSION TO THE VICINITY OF MARS USING PHOBOS AS A STAGING OUTPOST; SCHEMATIC VEHICLE DESIGNS CONSIDERING CHEMICAL AND NUCLEAR ELECTRIC

N89-17604

Unclas
G3/12 0189641

MARS STUDY GROUP MEMBERS

SEAN NOLAN
DEB NEUBEK
C.J. BAXMANN

MANNED PLANETARY VEHICLE CONCEPTUAL DESIGN
SEAN NOLAN

CREW COMMAND VEHICLE CONCEPTUAL DESIGN
DEB NEUBEK

SPACE FREIGHTER CONCEPTUAL DESIGN
C.J. BAXMANN

ACKNOWLEDGEMENTS

We greatly appreciate the guidance and technical assistance of Mr. Dennis Fielder

This study was produced with funding from the National Aeronautics and Space Administration - Universities Space Research Association Johnson Space Center, Houston, Texas

This is a student project completed by Master of Architecture candidates at the University of Houston Sasakawa International Center for Space Architecture, which required a substantial amount of preliminary research and coursework beyond the scope of this report

FOR FURTHER INFORMATION

Write to:
Sasakawa International Center for Space Architecture
Architecture-SICSA
University of Houston
Houston, Texas 77204-4431
713/749-1181

TABLE OF CONTENTS

TABLE OF FIGURES

LIST OF ACRONYMS & ABBREVIATIONS

1g	One Earth gravity
AMME	Advanced Manned Mars Explorer
CCV	Crew Command Vehicle
CG	Center of Gravity
EVA	Extra-Vehicular Activity
HLLV	Heavy Lift Launch Vehicle
ICTV	Interplanetary Cargo Transport Vehicle
IMME	Initial Manned Mars Explorer
LEO	Low Earth Orbit
LOTS	Lunar Oxygen Transportation System
MME	Manned Mars Explorer
MO	Mars Orbit
MPV	Manned Planetary Vehicle
OMV	Orbital Maneuvering Vehicle
OTV	Orbital Transfer Vehicle
PTF	Propellant Tank Farm
RMS	Remote Manipulator System
S1	Design Concept Scenario 1
S2	Design Concept Scenario 2
SDPG	Solar Dynamic Power Generator
SOC	Space Operations Center
SS	Space Station
STS	Space Transportation System (Space Shuttle)
rpm	Revolution Per Minute
kg	Kilogram
kw	Kilowatt
m	Meters
rpm	Revolution Per Minute
cm	Centimeter

Abstract

The Manned Mars Explorer (MME) project responds to the fundamental problems of sending human beings to Mars in a mission scenario and schematic vehicle designs.

The mission scenario targets an opposition class Venus inbound swingby for its trajectory with concentration on Phobos and/or Deimos as a staging base for initial and future Mars vicinity operations. Optional vehicles are presented as a comparison using nuclear electric power/propulsion technology.

A Manned Planetary Vehicle and Crew Command Vehicle are used to accomplish the targeted mission. The Manned Planetary Vehicle utilizes the mature technology of chemical propulsion combined with an advanced aerobrake, tether and pressurized environment system. The Crew Command Vehicle is the workhorse of the mission performing many different functions including a manned Mars landing, and Phobos rendezvous.

Introduction

The Manned Mars Explorer study had two primary objectives: 1) to develop a mission scenario to deliver a crew of six to the vicinity of Mars; and 2) to conceptualize a transportation system to accomplish this mission.

The mission scenario is developed around the concentration on Phobos and/or Deimos as the primary destination and consisted of the following: A Manned Planetary Vehicle (MPV) would be built in low Earth orbit (LEO), then outfitted with a crew of six. Using an opposition class Venus inbound swingby trajectory, the MPV would travel to the vicinity of Mars in approximately 300 days, where it would stay for 60 days before departing on the Venus inbound swingby leg to LEO requiring approximately 210 days.

The sixty day exploration period in the vicinity of Mars would consist of sending a crew of three to the surface of Mars for one week. The crew would then return to the MPV and spend the remainder of the time ferrying between the MPV and Phobos. During this period, they would perform scientific study on resource utilization of Phobos, and remote sensing of Mars.

The transportation system design encompasses several considerations, including: all chemical propulsion vs. nuclear electric propulsion, the issue of reliability vs. redundancy, the need for artificial gravity vs. zero gravity, and the use of necessary but undeveloped technologies such as large scale aerobraking and tether systems.

The primary components of the transportation system include a Manned Planetary Vehicle (MPV) and a Crew Command Vehicle (CCV). As part of an alternative split mission to Mars the design of an Interplanetary Cargo Transport Vehicle (ICTV) and Manned Planetary Vehicle (MPV) are presented, both using nuclear electric power.

A detailed comparison of chemical vs. nuclear electric propulsion was made for the MPV to determine the impact of these technologies on a MME. An all chemical mission was chosen as the most realistic for the first manned mission to the vicinity of Mars because of the mature level of technology and its established reliability.

The main components of the Manned Planetary Vehicle include the Power System; Pressurized Environment System; Structural System; Folding Aerobrake System; Four-Tether System; and Staged Propulsion System.

The MPV was designed to artificially create one Earth gravity (1g) for crew health and safety considerations. To accomplish this task a spinning vehicle concept was used, which required the use of a tether system. A tether system was conceptualized which resists twisting through a unique spreader system and four tether configuration. The tether would be deployed during trans-Mars coast and trans-Earth coast, and reeled in for all propulsive maneuvers.

Due to the long and dangerous nature of this mission, reliability of vehicle components was established as a driving force in design. This was shown through the design of a multi-functional Crew Command Vehicle (CCV) which would house the crew during all propulsive maneuvers, and also provide transportation for the crew between the MPV, Mars, Phobos and/or Deimos, and the LEO Space Station.

1.0 PROJECT OBJECTIVES

The overriding objective of the Manned Mars Explorer (MME) project is to study conceptual design options which are based upon background and assumptions outlined in sections 2 & 3. This report offers conceptual solutions for some of the most fundamental problems associated with a manned mission to the vicinity of Mars.

1.1 General

Justify a manned mission to the vicinity of Mars emphasizing scientific and industrial incentives.

Respond to the technical challenge of a Mars mission with regard to human factor related issues.

Concentrate on Phobos and/or Deimos as a natural space station of Mars, and resource base.

Research existing literature and organize a database.

1.2 Mission Planning

Develop a Macro Plan including precursor missions necessary for initial and successor MME's, and resulting infrastructure.

Develop a Mission Scenario including the selection of an orbital trajectory and activities performed during all phases of the mission.

1.3 Comparison of Chemical to Nuclear Electric Propulsion

Study an Advanced Manned Mars Explorer (AMME) split mission consisting of a Manned Planetary Vehicle (MPV) and an Interplanetary Cargo Transport Vehicle (ICTV), both utilizing nuclear electric propulsion. The purpose of this brief study is to assess the advantages and disadvantages of chemical and nuclear electric propulsion, and to serve as a comparison to the

Initial Manned Mars Explorer (IMME) mission which is the main focus of this study.

1.4 Development of Schematic Vehicle Designs and Transportation System

Conceptually Design an Initial Manned Mars Explorer (IMME) mission scenario and schematic vehicle designs including a Manned Planetary Vehicle (MPV) and Crew Command Vehicle (CCV). This mission is intended to be the main focus of the report.

2.0 BACKGROUND

2.1 Mission Incentives

The Mars Study Group researched possible political, scientific, and industrial incentives for going to Mars, recognizing the influence each group has in the Mars vicinity initiative. These basic incentives are expanded to include additional key incentives reletive to the goal of the MME mission.

International cooperation
Scientific information
Industrial/Economic resources
Phobos and/or Deimos resources
Pioneering spirit

International cooperation on a MME involves the sharing of the costs and benefits of such a program with other nations while extending the human presence to another part of the solar system. The tangible benefits (e.g., scientific, technological and economic) of this program are significant; however, the greatest spinoff will be an intangible: "Worldwide cooperation in space may produce increased worldwide cooperation on Earth" (Goldman 1985).

Scientific information is present in the vicinity of Mars that may help answer many questions about the formation and composition of the solar system, in addition to clues of the past and/or future of the Earth (Glass 1982; Singer 1986).

Industrial/Economic incentives exist in the vicinity of Mars based on the currently envisioned mineral composition of the surface and near surface environment of Mars, Phobos, and Deimos (Mutch et al. 1976). In addition, the atmosphere of Mars contains 1.6% Argon, and 2.7% Nitrogen (Glass 1982) which could be used in breathing gas. The potential for mining materials in the 0.38 g environment of Mars (Abell 1982) and/or the 0.0006 g

(Boston 1984) environment of Phobos and Deimos is also an important incentive. These resources could help support a LEO Space Station and/or Lunar Base given an economical and reliable transportation system between LEO and MO.

Phobos and/or Deimos have a tremendous resource potential. In general the potential uses and resources available on the Martian moons are:

Scientific information
Mars observation
Mineral resources
Propellant base
Construction materials
Less energy required to land
Short travel time to Mars
Resource stockpiling base

The above mentioned incentives are descriptive of the ideal spaceport. The overriding incentive for the development of Phobos as a spaceport is its .0006 g environment, making docking and stockpiling nearly effortless. In addition Phobos and Deimos have inferred composition similar to carbonaceous chondrites (Science 1978) which would provide valuable constituents for propellant and construction materials (O'Leary 1987). With its near vicinity to Mars (roughly the distance between Australia and the United States), Phobos could act as a stockpiling depot for materials traveling from the surface of Mars to LEO or Lunar Base. In addition, the moons of Mars are excellent vantage points for Mars observation.

The pioneering incentive for society to expand into space and open new frontiers is perhaps the least quantifiable and most important of all the above incentives. The people of the United States, and the World, have consistently overcome barriers in the name of exploration, and will collectively be the decision making constituency supporting planetary exploration.

2.2 Mission Objectives

For a successful MME it will be necessary to combine ambitious goals with achievable objectives, and be committed to long-term involvement. The main mission objectives are:

Learn more about Mars, its moons, and the evolution of our solar system.

Bring nations together politically.

Develop an efficient transportation and industrial infrastructure.

Concentrate on Phobos and/or Deimos:
Initially to explore potential resource applications.
Ultimately to exploit useful resources.

2.3 Manned Presence Justification

Manned presence offers the benefits of:
Intellect
Innovation
Intuition

These skills help in performing the following functions:
Troubleshooting complex problems.
Installing experiments.
Monitoring operations on site.
Prospecting for samples.

Increased human presence in space is a goal removed from scientific and technical discussions which identifies an aspiration for mankind in general.

2.4 The Issue of Reliability vs. Redundancy

The issue of reliability vs. redundancy is one which is inherent in the planning of a technically complex mission. Throughout the mission planning stage of this project as well as the concept design phase, industry experts were queried to determine what level of reliability could be established without unnecessary redundancy. The final concept designs presented in this report are based on the assumption that in a mission as long and dangerous as a manned Mars excursion, a highly reliable, fault tolerant transportation system must be in place. (Carr 1988).

The implications of this assumption are reflected throughout the concept designs, particularly in the Crew Command Vehicle which is discussed under section 4.0.

3.0 PROJECT ASSUMPTIONS

The Mars Study Group identified several project assumptions that may have a significant impact on a MME mission. The following assumptions are based on existing and evolving technologies, recognizing that significant technological advancements will be required for a practical MME mission.

3.1 LEO launch capacity

Low Earth orbit launch capacity required for delivery of MME components are:

Heavy Lift Launch Vehicle (HLLV) sized for heavier and larger payloads necessary for large pressurized modules, structural system components, propulsion stage components, and fuel. Baseline capacity = 181,000 kg (Page 1986).

Space Transportation System (STS) (popularly known as the Space Shuttle) for crew transfer and delivery of smaller and lighter payload. Space Shuttle Derived Vehicles may deliver as much as 82,500 kg. Baseline capacity = 29,000 kg (Page 1986).

3.2 LEO infrastructure

Low Earth orbit infrastructure required for the fabrication of MME components are:

Space Station (SS) is a permanently-manned and operational international endeavor providing crew-support functions-- in particular, life sciences research and studies to facilitate prolonged periods of productive living and working in space-- as well as serving as a technology testbed for life support systems, automated systems and robotics.

Propellant Tank Farm (PTF) in LEO provides storage for large quantities of propellants and an orbiting depot for refueling operations for Orbital Transfer Vehicles (OTVs) and other transportation systems requiring on-orbit fueling.

Space Operations Center (SOC) is a larger structure (compared to SS) in LEO essential in supporting advanced transportation operations with hangar and servicing facilities, advanced power systems, increased operations capabilities and additional habitation and research modules. Its main function is the on-orbit assembly of large transportation vehicles.

Orbital Transfer Vehicles (OTV) are the workhorses of the LEO-Moon infrastructure, providing transportation to and from various Earth and Lunar orbits in support of payload delivery systems as well as numerous manned and unmanned space operations (including vehicle assembly/staging and orbit-raising).

Orbital Maneuvering Vehicles (OMV) are reusable,tele-operated, free-flying vehicles used in LEO and in the vicinity of Mars for a variety of on-orbit services in support of orbiting elements (including ferrying equipment between co-orbiting elements and remote servicing operations).

Lunar Oxygen Transportation System (LOTS) operating between LEO and the Moon and serviced by a fleet of OTVs will provide the Earth-Lunar infrastructure with an important commodity of lunar development-- propellants-- by transporting significant amounts of oxygen from the lunar surface to LEO at substantial cost savings in comparison to delivery from Earth.

3.3 Technology Considerations

Current technologies presented in the MME, such as chemical propulsion, fuel cells, solar thermodynamic power, pressurized habitation volumes, and structural framing are considered near-term and could be delivered and operated in LEO within the next 5 - 7 years.

Evolving technologies presented in the MME, such as large scale aerobraking, large scale multiple tethers, rotating vehicle, and large scale propulsive maneuvers are considered to be advanced technologies requiring 10 -15 years for development.

3.4 Design Drivers

The MME project has identified key issues in the design of a manned mission to the vicinity of Mars. These issues are central to human factors and overall vehicle mass considerations:

Chemical propulsion was chosen over the option of nuclear electric propulsion for reasons described in section 4.0 Project Description.

Reliability is an important issue in inter-planetary travel. Due to the long and dangerous nature of this mission, and the enormous cost per pound of mass to perform the mission, the use of multi-functional, fault-tolerant, and reusable hardware must be a requirement.

One-Earth-gravity, as opposed to zero gravity, has been identified as a countermeasure to the medical maladies of long term exposure to the absence of gravity.

A rotating vehicle is presented as one method for providing the crew with artificial gravity for a majority of the mission.

All aerobrake Earth return offers an overall initial vehicle mass savings of approximately 50%. (See Appendix C Mars Propulsion System Assessment for comparison between all aerobrake and all propulsion scenarios.).

4.0 PROJECT DESCRIPTION

4.1 MME Macro Plan

The macro plan of the initiative to explore and exploit the vicinity of Mars is developed in its initial phases in preparation for a series of imminent manned missions and resulting infrastructure. The 50 year Mars vicinity initiative includes (in chronological order):

Precursor surveillance probes
Sample return probes
A Phobos (or) Deimos robotic mining and processing outpost
A manned mission to the vicinity of Mars
A Phobos and/or Deimos staging and resource base (Spaceport)
A Mars robotic mining and processing outpost
A cycling Earth-Moon-Mars transportation system
Mars as an industrial installation

Precursor Missions to the vicinity of Mars will supply the vast amount of information still needed on the atmosphere, geophysics, and geology of Mars, Phobos, and Deimos prior to any manned undertaking. The initiative to gather information will occur in incremental phases beginning with projects such as the U.S.S.R. Phobos probe planned for the early 1990's (Av. Wk. & Space Tech. 1987), and the U.S. Mars Observer (Ride 1987) planned for the late 1990's. It is important to note that in preparation for a manned mission a great deal more information will be required than for an unmanned mission. For this reason the length of time necessary for information gathering prior to a MME is subject to some uncertainty. Precursor missions will continue to be launched until mission planners are satisfied with their information.

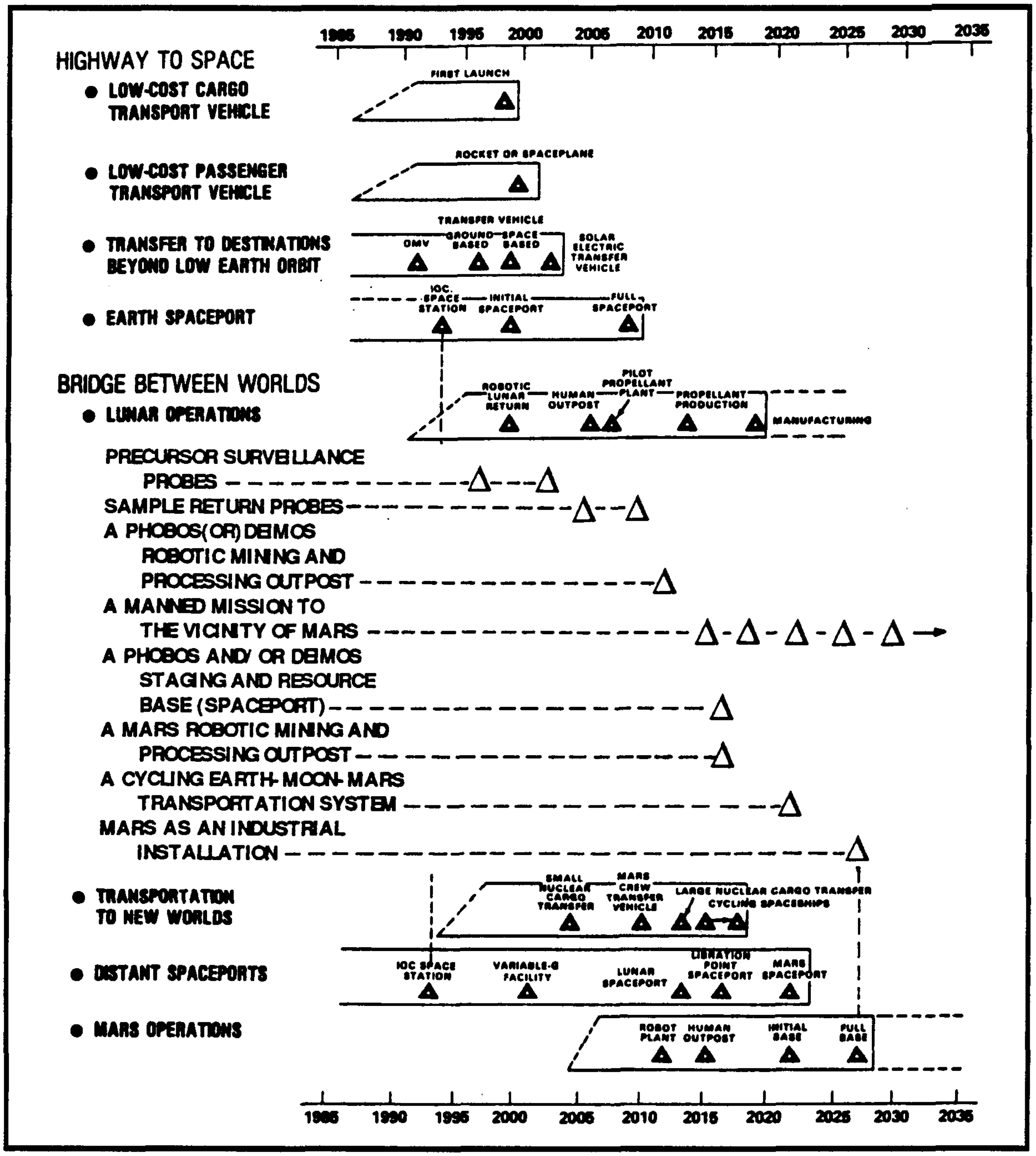


Figure 4-1 Macro Plan Timeline

The MME macro plan milestones have not been given specific dates; however, they are shown as they fit into the currently envisioned National Commission on Space timeline. This graphic shows the urgency with which we need to concentrate on these activities if a MME is to take place on schedule. Elements of the macro plan timeline are borrowed from Pioneering the Space Frontier (National Commission on Space 1986) to give context.

Sample return probes to Mars, Phobos and/or Deimos will follow with the main purpose of returning geologic and atmospheric samples from disparate sites on Mars and Phobos and/or Deimos to Earth for detailed analysis. The three year sample return missions would be most beneficial if launched in series with a time gap between them allowing scientists to analyze data from the first to plan experiments for the second.

An **Initial Phobos robotic outpost** would follow to test and evaluate on-site mining and processing. The moons of Mars offer an ideal site for this experiment due to their 0.0006 gravity environment.

A **manned mission to the vicinity of Mars** will follow to perform manned and man-tended scientific experiments on the surface of Mars, Phobos and/or Deimos, maintain the mining and processing equipment already in place, and install new elements of the Mars vicinity scientific and industrial infrastructure.

A **Phobos and/or Deimos staging and resource base (spaceport)** is envisioned as a depot for commodities traveling from MO to LEO. As a spaceport it could also be used as a staging base and refueling point for interplanetary vehicles traveling to the asteroids, Jupiter and its moons, and beyond.Phobos and Deimos are both natural Mars observation points.

A **Mars robotic mining and processing outpost** will be necessary to test materials mining and processing on the surface of Mars prior to a major industrial effort. The outpost may be delivered by a manned split mission or an unmanned Earth launch.

A **cycling Earth-Moon-Mars transportation infrastructure** should be capable of efficiently delivering raw materials to any point within the Earth-Moon-Mars cycle. Given the resources of Mars and Phobos it is conceivable that a cycling transportation system could deliver water, construction materials, atmospheric gasses, metals and propellant to LEO or a moon base (Toulmin, et al. 1977).

Mars as an industrial installation. The ultimate use of Mars will most likely be that of an industrial installation. A human colony would be in place for the sole purpose of tending to the operation and maintenance of the industrial installation. Assuming an industrial facility it may be desirable to have an entirely robotic (man-tended) installation to avoid the cost and danger of landing humans on the surface of Mars.

4.2 Analysis of Chemical and Electric Propulsion Scenarios

Comparison studies of chemical propulsion and electric propulsion were made to determine their applicability to the design problem at the outset of the design process. The major advantages and disadvantages of each system are:

Chemical propulsion advantages include (1) mature technology (no development cost, available now); (2) established performance and reliability; (3) high thrust (capable of launch from Earth and other bodies with high-gravitational fields); (4) short trip times (due to high thrust capabilities).

Chemical propulsion disadvantages include (1) high propellant consumption; (2) high propellant-to-payload weight ratio; (3) higher delivery cost (large propellant quantities delivered from Earth to LEO); (4) low specific impulse (Isp < 500 sec.).

Electric propulsion advantages include (1) low propellant consumption; (2) low propellant-to-payload weight ratio; (3) lower delivery cost (small propellant quantities delivered from Earth to LEO); (4) high specific impulse (Isp > 500 sec., up to 10,000+ sec.).

Electric propulsion disadvantages include (1) evolving technology (development cost required, not immediately available); (2) unproven performance and reliability; (3) low thrust (limited to missions in low-gravitational fields-- i.e., orbit-raising and maneuvering, trans-orbit operations); (4) long trip times (due to low thrust capabilities).

A follow-up schematic study of two transportation vehicles provided a comparison based on the overall mass delivery requirements to LEO. The results indicated a chemicallypowered vehicle mass of approximately 4,000,000 kg versus a nuclear electric vehicle mass of about 450,000 kg to perform the same mission with similar payloads. Figure 4-2 shows the LEO support infrastructure (SS, SOC, PTF) for the MME and the subsequent Mars transportation systems (AMME, IMME, ICTV) with representative STS and HLLV launch requirements based on vehicle mass assessments only.

A major design consideration for the MME is the utilization of only chemical propulsion to develop a refined chemical vehicle configuration which allows for some options for reducing overall vehicle mass. Although electric propulsion provides significant mass savings, the advantages of chemical propulsion's mature technology and established performance/reliability make the pursuit of the near-term goal possible.

In terms of future human exploration of Mars, recent reports by Ride (1987) and the National Commission on Space (1986) advocate a split-mission concept-- a "fast" (presumably chemical) personnel transport and a cargo vehicle which "minimizes its propellant requirements by taking a slow low-energy trip to Mars" and utilizes "efficient interplanetary propulsion" (Ride). To fulfill this need, a conceptual design study of an Interplanetary Cargo Transport Vehicle (ICTV) was conducted to identify the vehicle's mission needs and capabilities (see Appendix A ICTV Performance Summary). This effort was influenced by a previous study (Phillips 1987) and is briefly presented as a point of reference.

The ICTV, shown in Figure 4-3, is approximately 130 meters long and 25 meters wide and utilizes nuclear electric propulsion for transporting large payloads from LEO to low Mars orbit in a circular spiral trajectory. Power is provided by a 3 megawatt electric nuclear power source based on the SP-100 Nuclear Power System currently under development by NASA and others. The reactor and shield are located at the front of the vehicle just ahead of the conical radiator. At the other end of the spacecraft is the thruster module with 50-cm. xenon ion thrusters for propulsion. The 5-m. erectable beam structure is deployed as a "spine" with three support masts for flexibility both in payload attachment and vehicle configuration. Two mobile remote manipulator systems (MRMS) operate along the spine, and chemical reaction control thrusters (RCS) are mounted on the truss structure for attitude control.

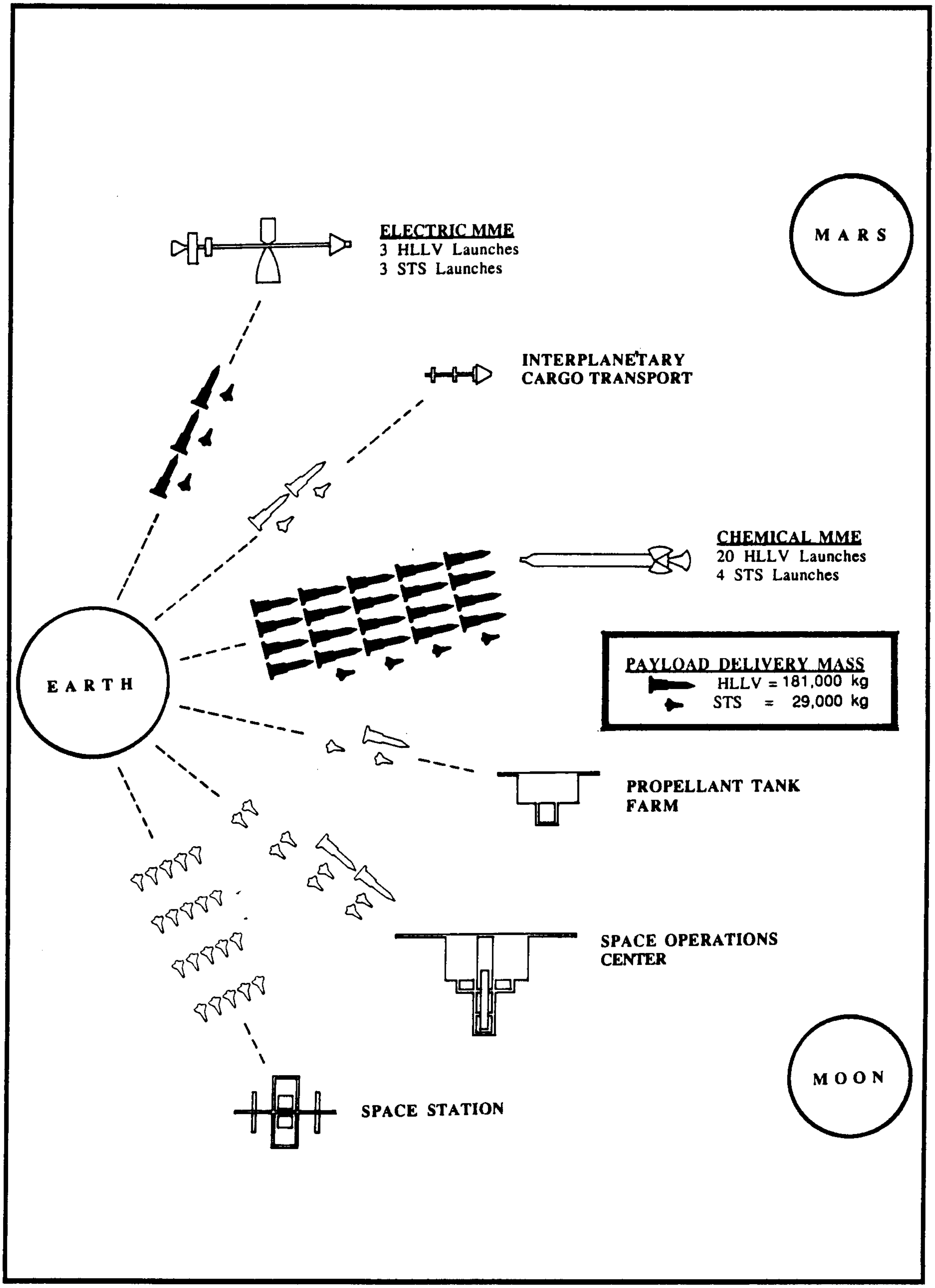


Figure 4-2 LEO Infrastructure with Representative Launch Req'ts.

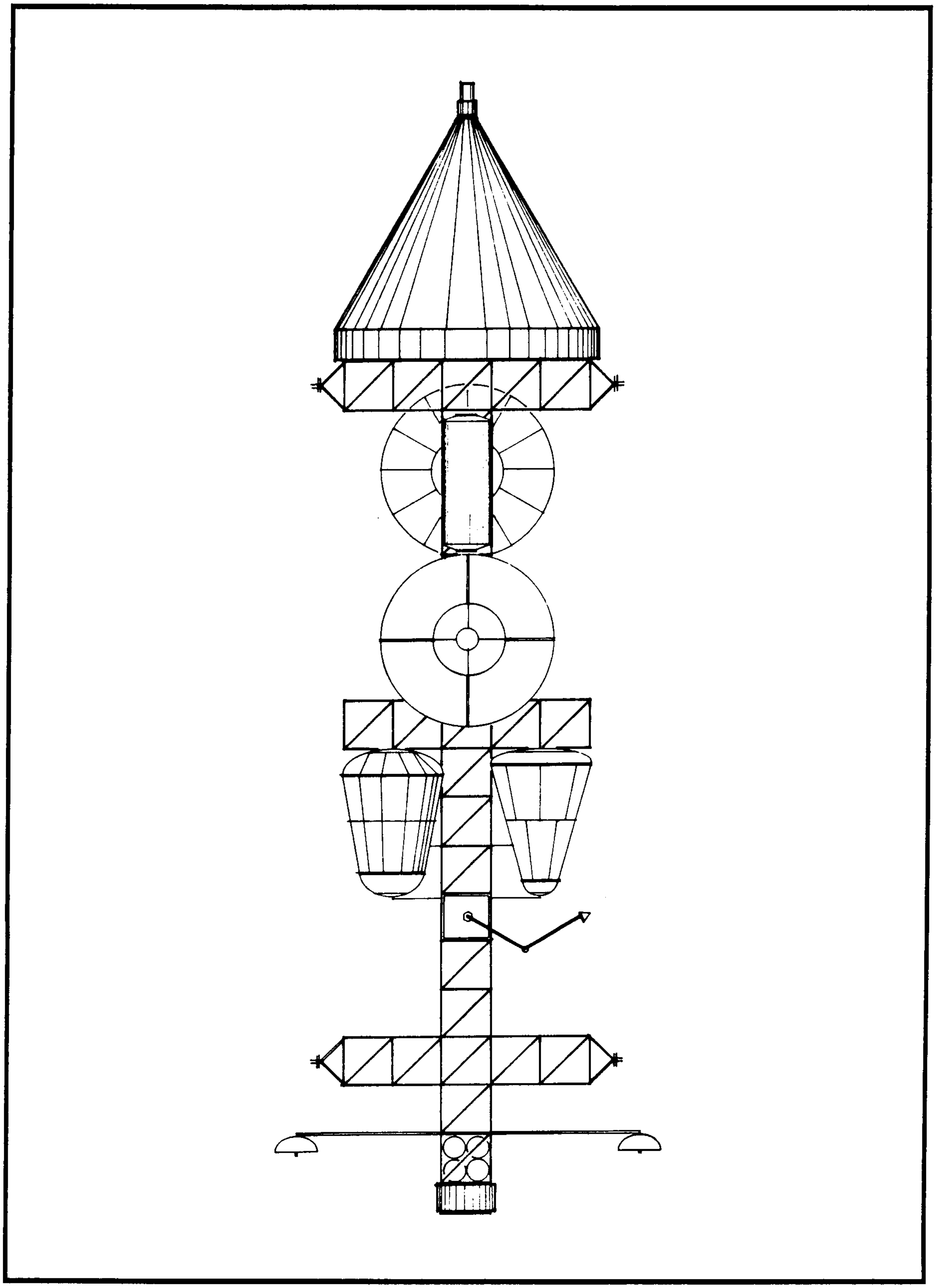


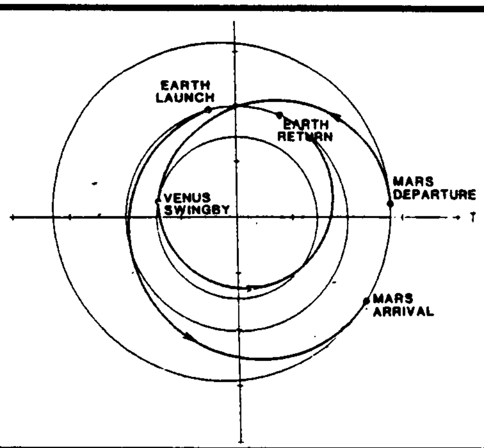
Figure 4-3 ICTV Concept Drawing.

4.3 Initial Manned Mission to the Vicinity of Mars

4.3.1 **The mission scenario** was developed after the following set of criteria/events were determined to be optimum:

Use of an opposition class Venus inbound swingby trajectory
Performing a manned Mars landing
Performing manned Phobos and/or Deimos landings

An **opposition class Venus inbound swingby trajectory** was chosen due to the relatively short overall mission length and stay time. The 60 day stay time allows a comfortable period of time for Mars and Phobos and/or Deimos exploration, prospecting, and resource evaluation. Optional scenarios which include stay time in the vicinity of Mars include conjunction class missions. These missions are characteristically longer in overall duration and require less energy than opposition class missions. Another trajectory option described by Hoffman (1986) as an up/down escalator offer advantages in the context of a cycling transportation system. This option was not suitable for the MME. Overall mission time was the deciding factor in favor of the opposition class mission given crew health and life support considerations.



Outbound leg = +/- 307 days
Stay = 60 days
Inbound leg = +/- 210 days
Total mission = 377 days
Opportunities every 26 months
Cyclic pattern repeats every 15 years

Figure 4-4 Opposition Class Venus Inbound Swingby Trajectory
Opposition class trip times are averaged from all opposition class Venus inbound swingby opportunities between January 2001 and November 2026 (Young 1986).

A manned Mars landing will put new experiments into place, maintain experiments already there, and prospect for new resources.

Manned Phobos and/or Deimos landings will set up an initial staging base for Mars vicinity activity. Phobos and Deimos offer nearly the same incentives for their utilization which include: mineral resources, a negligible gravity environment, and Mars observation capability. Many excursions will be planned for the surface of Phobos and/or Deimos after the manned Mars landing.

4.3.2 The mission phases were developed based on the preceding assumptions and consist of the following activities:

1. **Low Earth Orbit construction**
   a. Vehicle assembly
   b. Crew training
2. **Trans-Mars injection**
   a. Propulsive maneuver
   b. Communication satellite deployment
   c. Spin-up
   d. Power system deployment
   e. Tether system deployment
   f. Trans-Mars coast
   g. De-spin
   h. Power system retrieval
   i. Tether system retrieval
   j. Communication satellite retrieval
3. **Mars circularization**
   a. Propulsive maneuver
   b. CCV surface operations
   c. CCV return to MPV
4. **Trans-Earth injection**
   a. Propulsive maneuver
   b. Communication satellite deployment
   c. Spin-up
   d. Power system deployment
   e. Tether system deployment
   f. Trans-Earth coast
   g. De-spin
   h. Power system retrieval
   i. Tether system retrieval
   j. Communication satellite retrieval
5. **Earth orbit capture**
   a. Propulsion stage, CCV, and MPV separation
   b. Propulsion stage remains in hyperbolic orbit
   c. CCV propulsively circularizes at LEO with crew
   d. MPV aerobrakes into SOC orbit

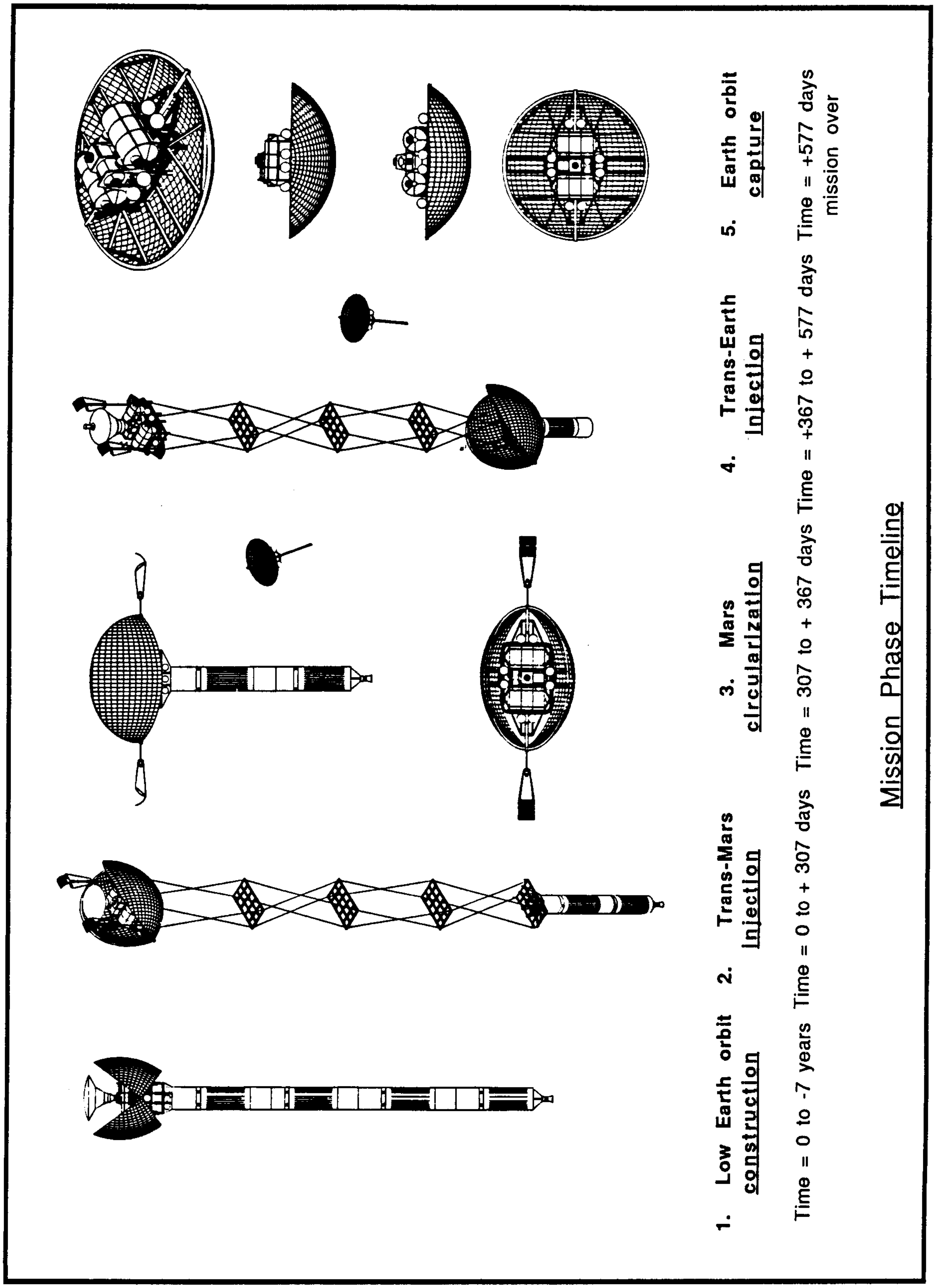


Figure 4-5 Mission Phases

Low Earth Orbit construction of the Manned Planetary Vehicle (MPV) using a Space Operations Center (SOC) for construction, and a Propellant Tank Farm for stockpiling propellant. The six person crew will be trained during this phase of the mission.

Trans-Mars injection propulsive maneuver will accelerate the MPV into a hyperbolic trajectory toward Mars for an outbound leg of +/- 307 days.

The communication satellite will deploy and precede the MPV at a distance that will allow communication using a low power wide bandwidth omnidirectional antenna. The communication satellite will use a high power narrow bandwidth parabolic antenna to send and receive signals from Earth.

Spin-up will begin by firing the reaction control system (RCS) thrusters which are located at various points along the propulsion stage and on the folded aerobrake. The vehicle will slowly begin spinning about its natural CG.

The power system and tether system will begin to deploy when the MPV reaches approximately 0.25 rpm. The slow rotation will help the systems deploy.

Trans-Mars coast will begin when the tether system deploys to a length of approximately 450 feet at which time a 2 rpm cycle will begin, artificially simulating 1g in the pressurized environment. During this mission phase the crew will engage in life sciences experiments, astronomical experiments, and training activities.

De-spin will occur in advance of Mars vicinity arrival to allow for any necessary course correction. The RCS system will slow MPV rotation to approximately 0.25 rpm.

Power system and tether system retrieval will occur once the MPV has slowed to approximately 0.25 rpm and the auxiliary power system is on-line.

Communication satellite retrieval will occur once the MPV has stopped rotating and is ready for docking maneuvers.

Mars circularization will occur when all systems are stowed and the crew members are safely seated in the CCV. The propulsion system will perform a propulsive braking maneuver circularizing into a parking orbit between Mars and Phobos.

The CCV will perform surface operations on Mars, Phobos and/or Deimos. The 60 day exploration period consists of sending a crew of three to the surface of Mars for one week in the CCV. After the crew has returned from the surface of Mars a three man crew will make several excursions to the surface of Phobos and/or Deimos. The CCV will then return the crew to the MPV for the next mission phase.

Trans-Earth injection propulsive maneuver accelerating into a hyperbolic Venus inbound swingby trajectory toward Earth lasting +/- 210 days.

The communication satellite will deploy and precede the MPV at a distance that will allow communication using a low power wide bandwidth omnidirectional antenna. The communication satellite will use a high power narrow bandwidth parabolic antenna to send and receive signals from Earth.

Spin-up will begin by firing the reaction control system (RCS) thrusters which are located at various points along the propulsion stage and on the folded aerobrake. The vehicle will slowly begin spinning about its natural CG.

The power system and tether system will begin to deploy when the MPV reaches approximately 0.25 rpm. The slow rotation will help the systems deploy.

Trans-Earth coast will begin when the tether system deploys to a length of approximately 650 feet at which time a 2 rpm cycle will begin, artificially simulating 1g in the pressurized environment. During this mission phase the crew will engage in life sciences experiments, astronomical experiments, and training activities in preparation for Earth orbit re-entry.

De-spin will occur in advance of Earth vicinity arrival to allow for any necessary course correction. The RCS system will slow MPV rotation to approximately 0.25 rpm.

Power system and tether system retrieval will occur once the MPV has slowed to approximately 0.25 rpm and the auxiliary power system is on-line.

Communication satellite retrieval will occur once the MPV has stopped rotating and is ready to receive it.

Earth orbit aerobrake/crew re-entry. Upon Earth arrival the crew will enter the Crew Command Vehicle (CCV) before the propulsion stage separates from the MPV. After separation the CCV will perform a propulsive maneuver to circularize Earth into LEO with its remaining fuel. Shortly thereafter the MPV will perform an all aerobraking maneuver with a propulsive assist to raise perigee and circularize into the orbit of the Space Construction Post.

4.4 Transportation System

The transportation system conceptual design encompassed several considerations as outlined in section 3.0 project assumptions. This conceptual design illustrates the combination of mature technology combined with advanced technology to offer alternatives for a cost efficient, humanly practical mission to the vicinity of Mars.

The primary components of the transportation system include:

A Crew Command Vehicle (CCV)
A Manned Planetary Vehicle (MPV)

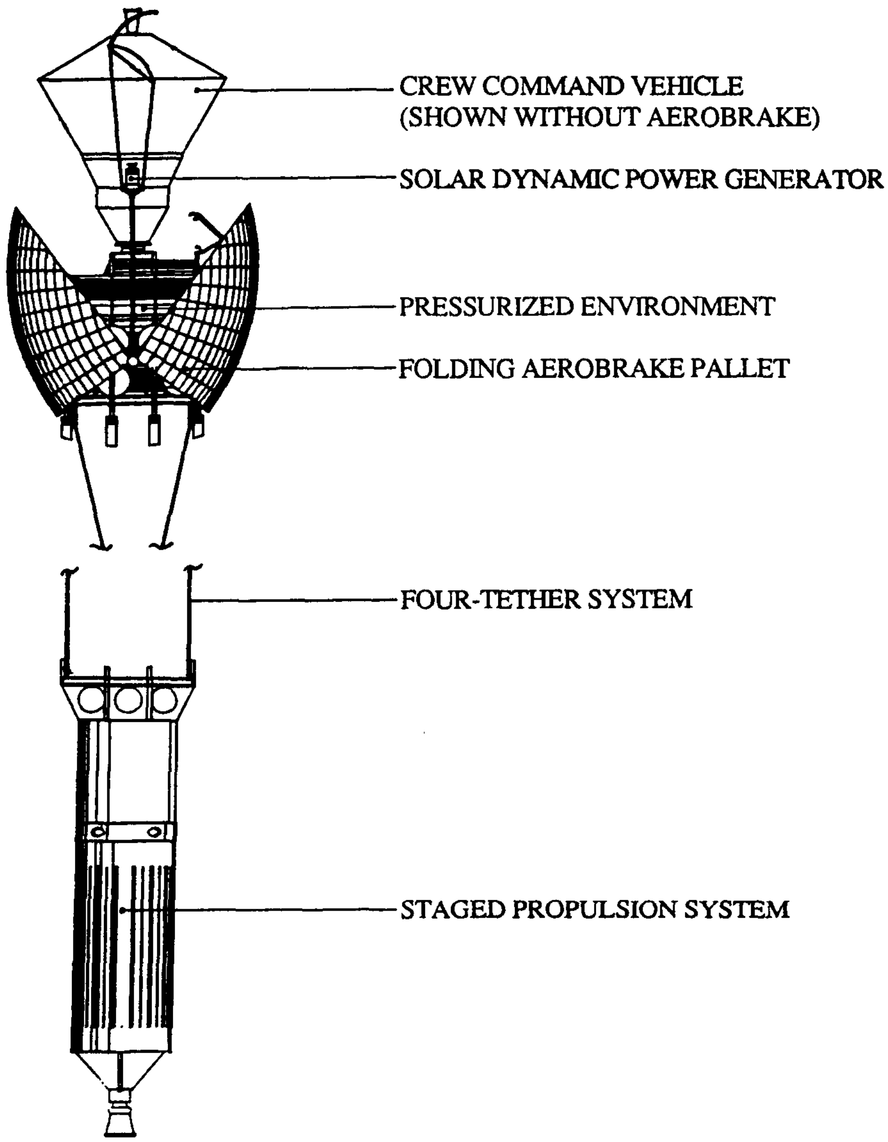


Figure 4-6 MPV Main Components

4.5 Crew Command Vehicle

The Crew Command Vehicle (CCV) best illustrates the value of reliable and reusable components. Due to the long and dangerous nature of this mission, and the enormous cost per pound of mass to perform the mission, the use of multi-functional, fault-tolerant, and reusable hardware must be a requirement. Therefore a CCV was conceptually designed which could accomplish the following primary functions:

Propulsive Maneuvers. The CCV will be occupied by the crew during all propulsive maneuvers required during the mission by the MPV.

Crew Transport. In the vicinity of Mars, it will be used to land a crew of three on the surface of Mars for one week of exploration and observation, then return the crew to the MPV. The CCV will then serve as a "ferry" to transport a crew to and from the surface of Phobos and/or Deimos.

Earth Orbit Return. In the vicinity of Earth, the CCV will be used to transport all six crew members to a Space Station orbit by detaching from the MPV and propulsively returning to Earth orbit.

As indicated by the above functions, the CCV is the workhorse of the transportation system and therefore critical to its success. It is assumed that it will be necessary for vehicles to perform such a variety of functions to maintain a practical mass limit for inter-planetary missions. The option of having a "spare" CCV onboard was researched but finally thrown out for this initial mission to Mars due to weight and reasonable reliability considerations. (Carr 1988). It is recommended that for advanced split missions, a "spare" crew return vehicle might be included in the manifest of the Interplanetary Cargo Transport Vehicle.

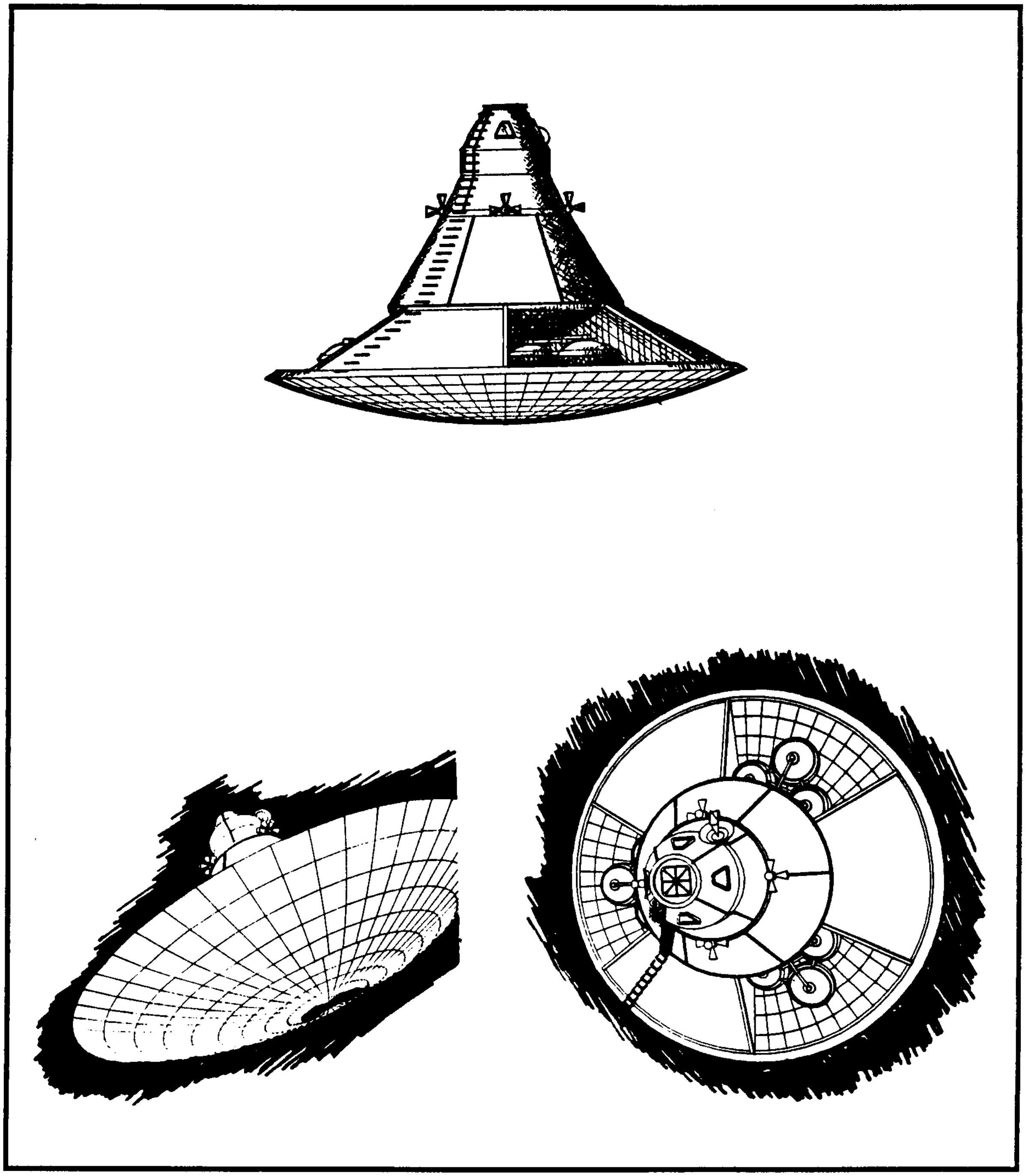
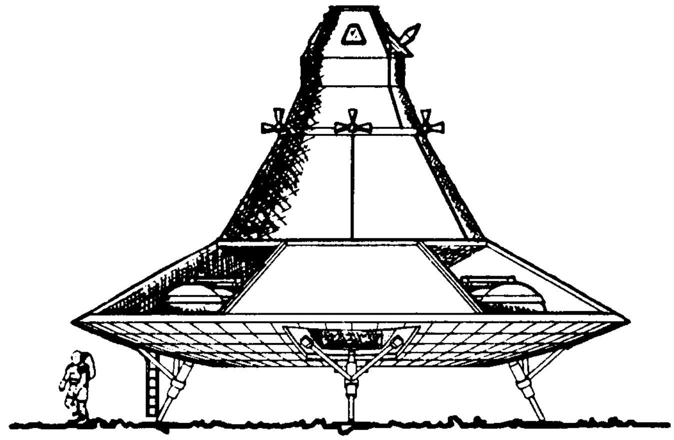
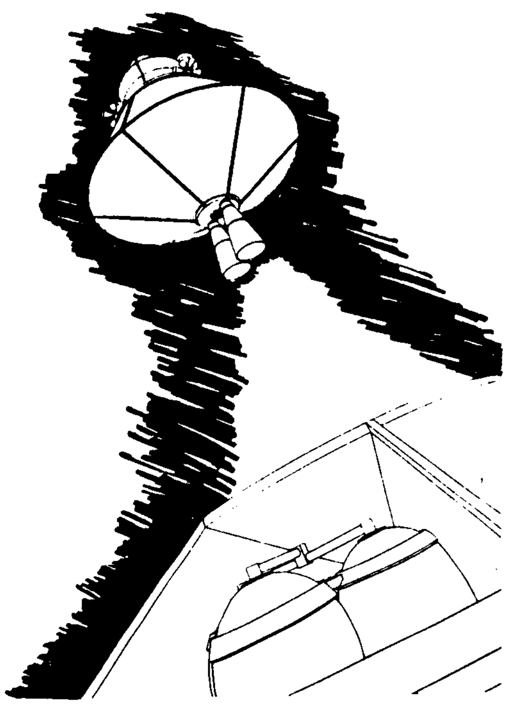


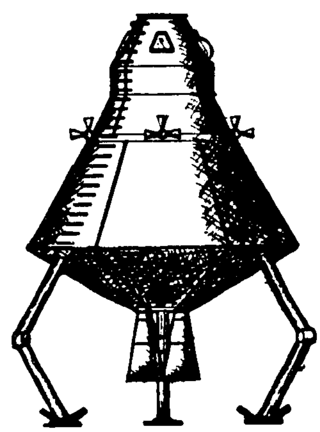
Figure 4-7 CCV During Mars Descent



On Mars Surface



Blastoff from Mars



Phobos/Deimos Configuration

Figure 4-8 Mars Vicinity CCV Configurations

TRANS-MARS INJECTION: Command Center; Escape Vehicle; G-Force Protection

MARS VICINITY:

MPV to Mars - Crew Transfer, Local Command Center, Science, Living Quarters

Mars to MPV - Crew Transfer, Local Command Center, Sample Return

MPV to Phobos - Crew Transfer, Local Command Center, Science/Sample Return, Living Quarters

Phobos to MPV - Crew Transfer, Local Command Center, Sample Return

TRANS-EARTH INJECTION: Command Center; G-Force Protection

EARTH ORBIT RETURN: Crew Transfer; Local Command Center; G-Force Protection

Figure 4-9 CCV Functions During Key Mission Events

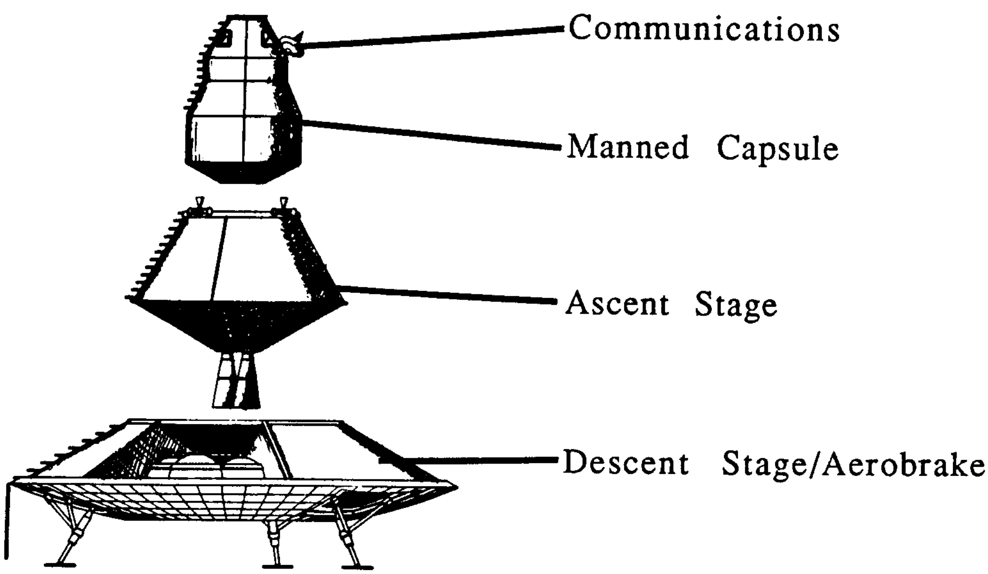


Figure 4-10 CCV Main Components

Crew Command Vehicle Components

The scope of this project included the conceptualization of a vehicle which could perform the many functions listed above. This led to the identification of primary components and target weights which may serve as a starting point for further engineering and study. Following are illustrations and descriptions of these components.

Manned Capsule. The manned capsule is a pressurized area sized for the accommodation of up to six crew members. It serves to protect the crew from g forces incurred during propulsive and aerobraking maneuvers and serves as command center, living quarters and safe haven during excursions. Its proportions are based on Orbiter cargo bay limitations. The scope of this project did not dictate an interior layout; however, several considerations were identified which included: design of a stacked, rotating couch/workstation required for proper "eyeballs in" crew orientation during the propulsive and braking maneuvers; and optimum placement of equipment/supplies to provide protection from radiation (Root 1965; Letaw and Clearwater 1986; Grandjean 1987).

Communications. A communications system has been provided to allow the crew to contact the MPV and serve as command center backup. A tracking dish is connected directly to the manned capsule/ascent stage.

Ascent Stage. The ascent stage consists of a propulsion system which is sized to take the crew and science payload back to the MPV. In addition to this, a landing/anchor system within the ascent stage was identified as a critical area for further study to be used when "docking" the CCV with Phobos or Deimos. The gear shown in Figure 4-8 illustrates the need for some type of removable anchor system in the landing pads.

Descent Stage/Aerobrake. The descent stage serves many functions. Primarily, it is used to slow the CCV during descent using an aerobrake and propulsion combination. A candidate system for the aerobrake thermal protection system such as that studied by General Dynamics under NASA contract 17085587 was used as baseline for weight and performance predictions. A rigid aerobrake is proposed which could incorporate removable sections that would be ejected to expose landing gear and descent engines for the final stages of descent. In addition, science storage areas would be provided within the aerobrake shell.

A weight budget of slightly more than 50,000 kg has been allocated for the CCV to perform the various missions described. A detailed weight summary may be found in Appendix b Crew Command Vehicle Weight Summary (Fielder 1988).

4.6 Manned Planetary Vehicle Components

The main function of the MPV is to provide transportation, habitation, and life support for a crew of six during the MME mission. In addition, the MPV will artificially simulate 1g during the outbound and inbound leg of the mission.

The main components of the Manned Planetary Vehicle include the Power System; Pressurized Environment System; Structural System; Folding Aerobrake System; Four-Tether System; and Staged Propulsion System.

4.6.1 Power System

A power system consisting of solar dynamic power generators and fuel cells was chosen as a possible concept to satisfy vehicle power needs. The solar dynamic power generators would be deployed during the rotation cycles, with the fuel cells being used during propulsive maneuvers and when solar power generation is not possible.

Solar Dynamic Power System with sun pointing capability is sized at 128 kg/kw (Sprengel 1987). The system is deployed when the MPV begins its rotation cycle and is immediately pointed toward the sun. The illustrated orientation of the solar dynamic power system was arrived at after an analysis of the MPV's orbital path, and rotation cycle. A constant orientation toward the sun is possible with very little pointing effort.

Fuel Cells are required for periods when the solar power system is not operational (e.g. planetary eclipse, propulsive maneuvers). The system is rated at 5 kg/kw (Rudey, R., et al. 1987).

Support systems for the fuel cells and solar dynamic power system are included in the mass estimates. These systems include collectors, concentrators, receiver/storage heat pipes, heat engine, radiator, power conditioning, and cabling.

Total system output 150 kw constant power.

Total system weight = 20,700 kg.

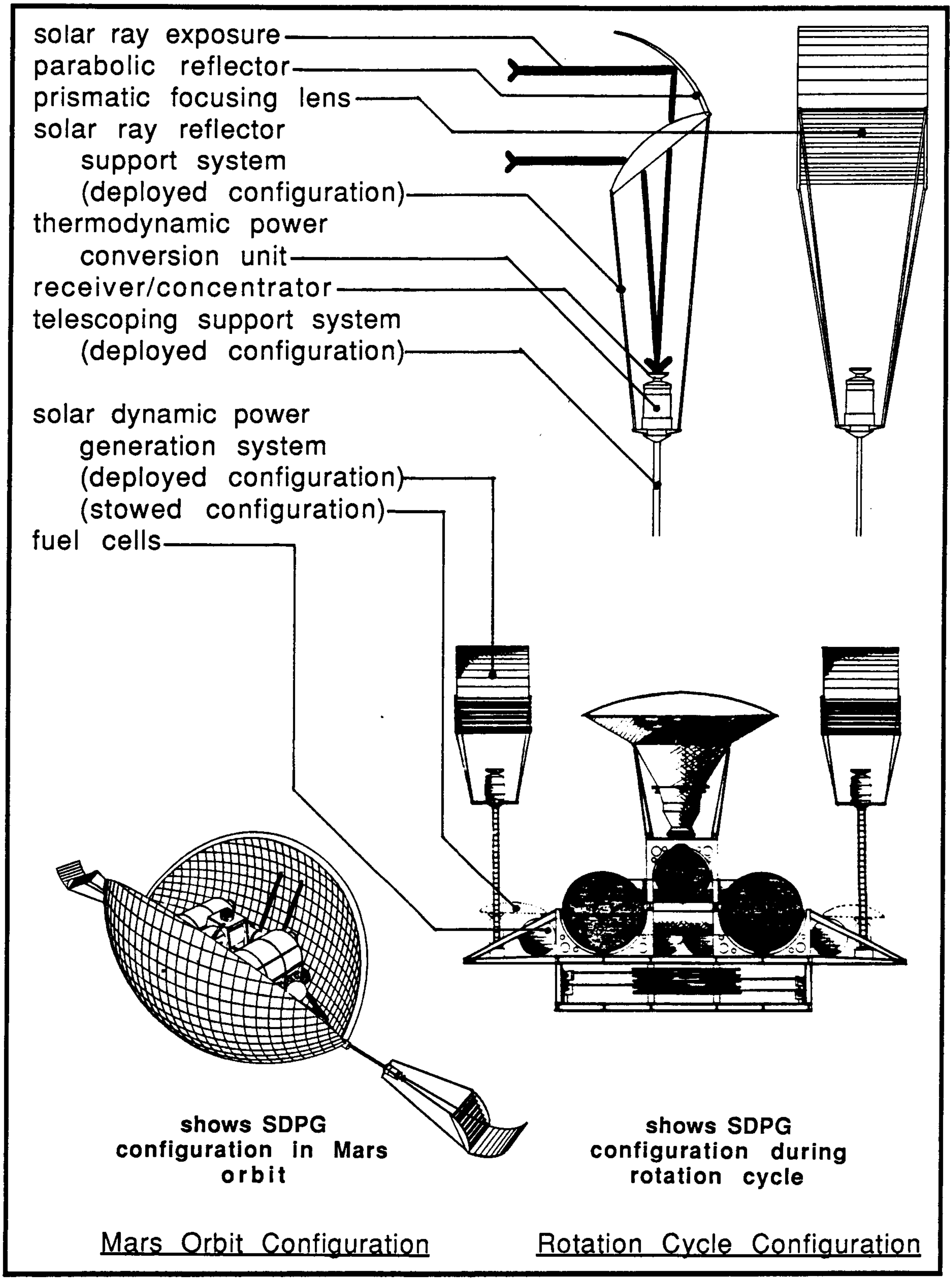


Figure 4-11 MPV Power System

4.6.2 Pressurized Environment System

The crew will occupy the pressurized environment system for the duration of the rotation cycles to and from Mars. The pressurized environment consists of a habitation module, laboratory module, safe haven, and connecting tunnels.

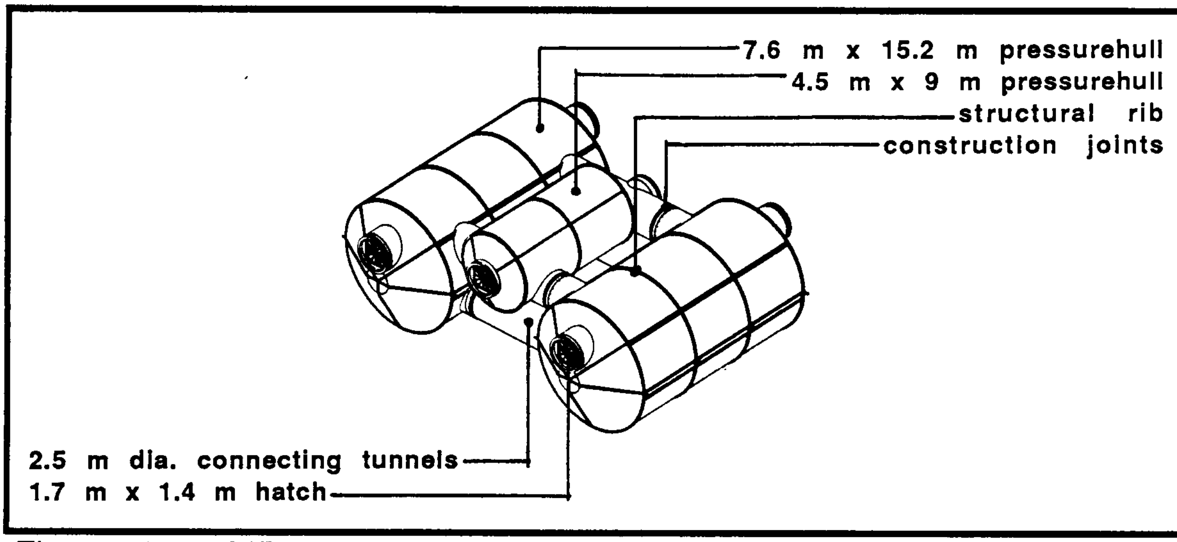
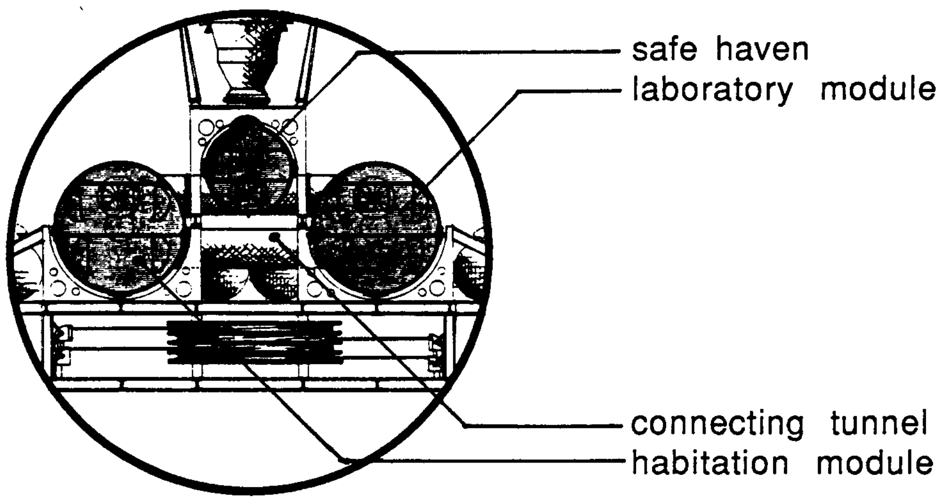


Figure 4-12 MPVPressurized Environment System

Habitation & laboratory modules are sized (7.6 x 15.2 meters) based on the envisioned capacity of a HLLV. Space station design studies will help in the layout of functions, color coding, crew interation, and general operation of a space station. The space station has little commonality on the structural design of the rack and floor system due to the 1 g loads parallel to the longitudinal axis of the pressurehull. Ergonomics will be very different in a 1g space station due to the reduced amount of easily accessible space.

Each large module will have three airlock sections, each with two means of egress; one to another pressurized airlock section, and the other to either the exterior or another airlock section. EVA equipment will be stored near each exterior egress used in case of planned EVA or emergency escape to the CCV.

Radiation protection in the large modules will be based on the average roetegen equivalent man exposure (rem) limit for astronauts (see Appendix D for radiation exposure constraints).



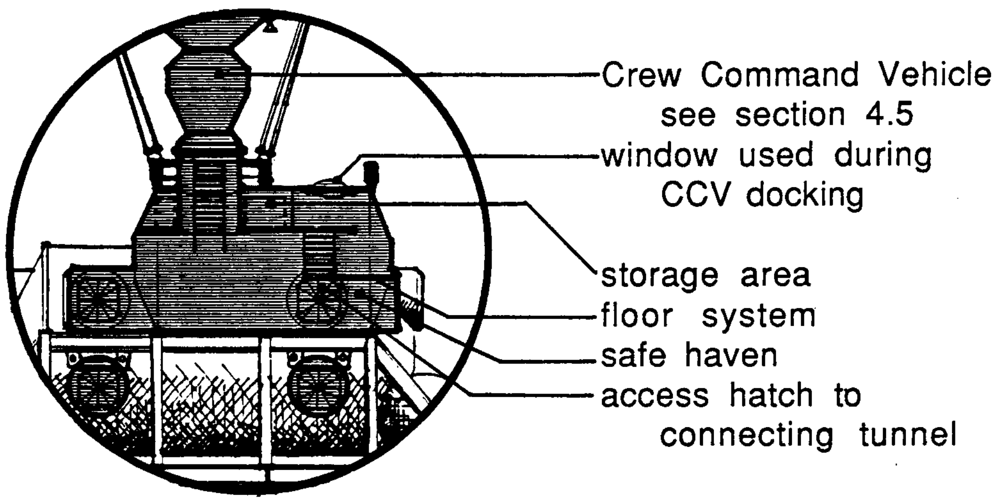
Section Detail Through Pressurehulls

Figure 4-13 Pressurized Environment Detail

Safe haven is sized (4.5 x 9 meters) based on the current STS payload bay. The safe haven will accommodate a crew of six during intense solar particle activity and dangerous maintenance operations. The actual length of solar particle events is subject to debate which leads to some uncertainty as to the appropriate design occupancy time of a safe haven. For the purposes of the MME 12 days was assumed. Solar particle events may be predictable in their seasonal intensity which would allow for planning the mission during a period of low solar activity (Rose 1987).

The safe haven will have one direct exterior egress, and another indirect exterior egress through the transportation tunnel. The transportation tunnel also provides one access to both large modules.

Radiation protection in the safe haven consists of aluminum shielding with sufficient mass (approx. 10 cm thick) (Letaw 1986) to dissipate ionizing radiation for the maximum anticipated solar particle event.



Section Detail Through Safe Haven and CCV

Figure 4-14 Pressurized Environment Detail

Connecting tunnels will allow transportation between the pressurized modules. The tunnels run in a "racetrack" configuration on both levels. This allows easy access to any point within the pressurized environment system in addition to providing safe egress to other pressurized areas in case of emergency.

Radiation protection in the connecting tunnels in minimal assuming little human occupancy.

EVA activity. Current designs for EVA suits include the AX-5 all-hard metal suit and the zero breathable suit Mk. 3 hard and soft suit (Av. Wk. & Space Tech. 1988). A modification of these suits may be necessary for prolonged solar radiation exposure during a MME. An exterior deck and handrail system will accommodate movement around the exterior of the MPV for use during zero or 1g operation.

Total system weight budget = 98,000 kg.

4.6.3 Structural System

The Structural System is designed to withstand forces generated by 3.5 g propulsive maneuvers and 1 g rotation cycles. The acceleration limit was established for crew health and structural considerations (Carr 1988).

The conceptual design is based on structural steel framing principles used for heavy equipment. The modular design allows easy installation and removal of the pressurehulls from the structural frame in case of modifications before or after the mission. The layout of the structural system, as it reflects the pressurized environment layout, allows optimum thrusting through the center of gravity. Load calculations and engineering studies have not been performed on the illustrated design.

Candidate materials consist of steel combined with various metal alloys, and possibly graphite or carbon additives.

Total system weight budget = 32,000 kg.

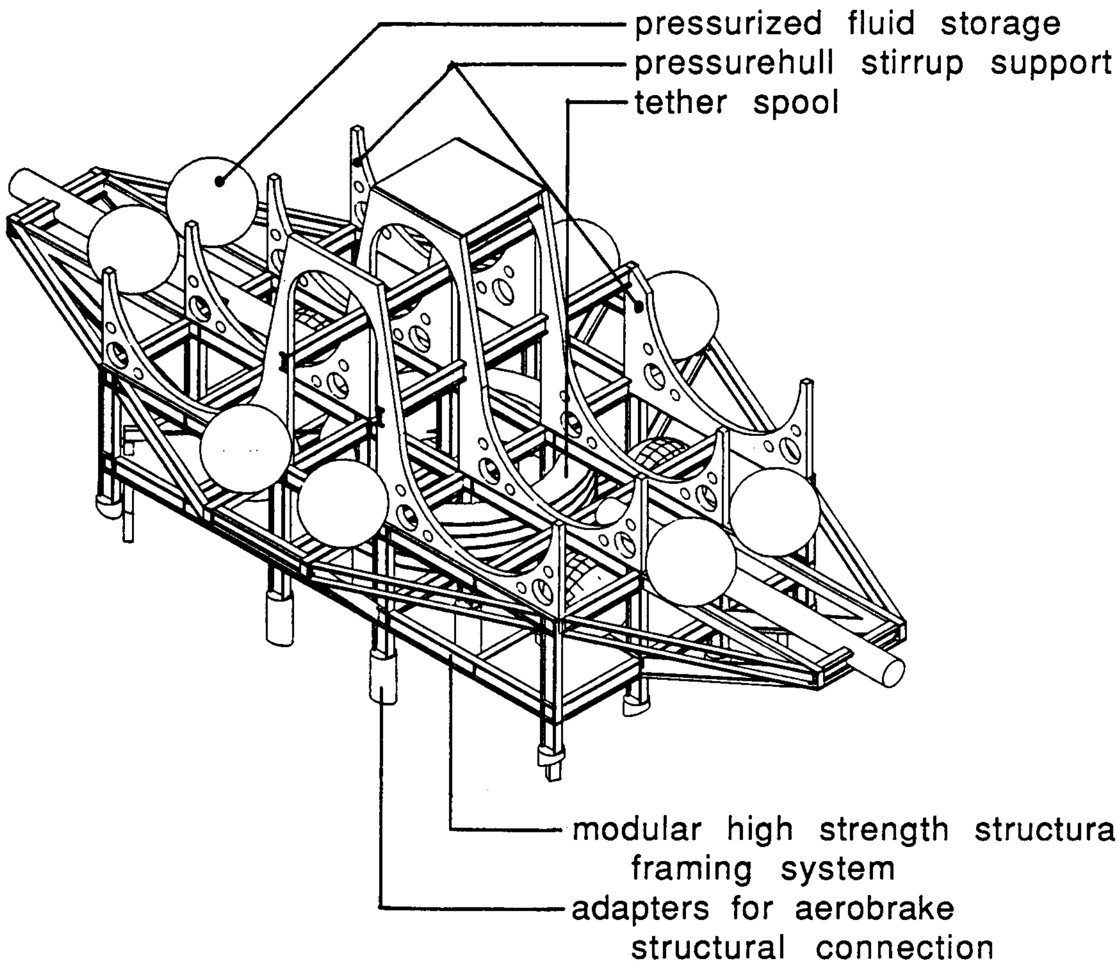


Figure 4-15 MPV Structural Frame

4.6.4 Folding Aerobrake System

The main function of the folding aerobrake system is to allow the power system, pressurized environment system, structural system, and tether system to circularize into the orbit of the SOC at a substantial overall mission propellant savings of approximately 50%. This was established as a desirable design option due to the reuse vale of the above mentioned subsystems (see Appendix C Mars Propulsion System Assessment).

The Folding Aerobrake System consists of the aerobrake in two folding sections, a transferrable structural pallet, folding mechanism, and fuel for earth circularization. The aerobrake is also used as a movable counterweight mass for the rotation cycles.

Open configuration allows easy operation of tether system during spin-up and retrieval. In this configuration the aerobrake also provides nominal protection from direct solar radiation (this was not a design requirement).

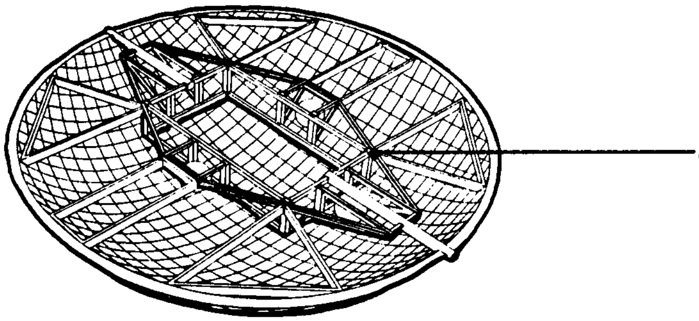
Closed configuration allows the above mentioned subsystems to use friction supplied by Earth's dense atmosphere of between 10^{-4} kg/m^3 to 10^{-12} kg/m^3; between 45.2 nm and 350 nm altitude (Dauro 1986).

Aerobrake Pallet supports folding apparatus, and propellant for Earth circularization and rendezvous with the SOC.

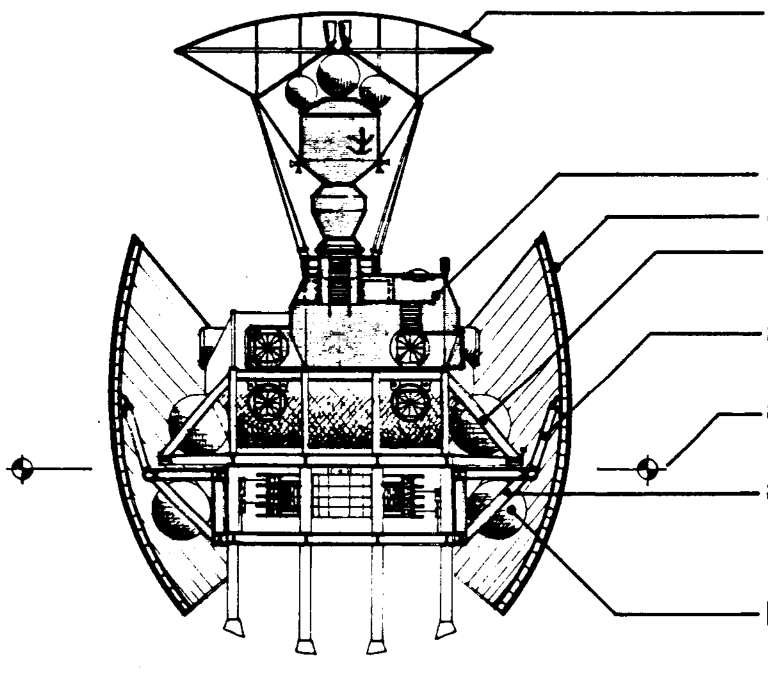
The Aerobrake consists of layered honeycomb structure providing heat insulation and structural support, covered with flexible heat rigidized material (General Dynamics; Martin Marietta).

A schematic failure analysis of the aerobrake system indicates that if the folding aerobrake was not functional, the MPV components would remain in a hyperbolic orbit for possible future retrieval. No loss of life would occur as the crew would re-enter in the CCV as per normal operating procedure.

Total system weight budget = 70,000 kg.



Aerobrake Open Configuration



Aerobrake Closed Configuration

Figure 4-16 MPV Folding Aerobrake System

4.6.5 Four-Tether System

A tether system was identified by the Mars Study Group as an optional subsystem for a spinning vehicle. A tether was chosen over a rigid or folding boom for weight and logistics considerations. Since the spinning vehicle is man-rated and will require reasonably accurate pointing ability and stability control, a four-tether system was conceptualized which may nearly eliminate detrimental oscillation and perturbations encountered during rotation cycles. A small scale concept model was constructed which satisfies the above hypothesis enough to warrant further engineering study of the tether system's dynamic modes.

Tether length is directly proportional to the changing mass of the propulsion system and the rate of rotation. For the purposes of this study the rotation speed was fixed at 2 rpm; and tether length was calculated based on the starting propulsion system mass of the two rotation cycles. In reality the rotation speed will most likely increase to compensate for the decreasing propellant mass, as the tether system may be locked into place for the duration of the rotation cycle for safety reasons.

The tethers are of sufficient length to provide one Earth gravity at two rpm during each rotation cycle.

Candidate tether specification:

DuPont aramid 49 fiber offers a tensile strength of 2758 MPa (400,000 psi)...Design diameter = 1 cm based on a oversized load of 320,000 kg. Solar radiation, bending radius and impact considerations may require a minimum diameter of 5 cm for each tether (Du Pont 1983). See Appendix F for further tether design data.

Tether system components may consist of the following:

Geared and motorized take-up spools will reel in tethers separately or at the same time...9 m diameter reduces the chance of tether shape deformation.

Pulleys and guides feed the tether from the spool, through the structural system, and to the counterweight.

Spreaders separate the tethers and contribute to the stability of the system.

Vibration/oscillation dampening mechanism may contribute to reduce perturbations in the system.

Design concept scenarios were used to analyze the ramifications of different design options. The scenarios are based on conceptual design options which may impact the overall vehicle delivery mass to LEO, and as a result, the length of the tether system.

The design concept scenarios also represent a range of velocity increments that are possible due to the orbital mechanics of different opposition trajectories as described by Babb (1986).

During the conceptual design process it was recognized that aerobraking the MPV at Earth left little counterweight mass for the return rotation cycle. This observation led to the conceptualization of two design concept scenarios which are described as follows:

Design concept scenario 1 hereafter referred to as "**S1**" assumes the following set of circumstances:

Total mission velocity increment = 10,475 m/sec.
Trans-Earth injection propulsion stage containment mass is retained for counterweight mass on return leg rotation cycle.
Aerobrake is transferred from MPV to propulsion stage for counterweight mass on return leg rotation cycle.
Trans-Mars injection separated into two propulsive maneuvers.

Design concept scenario 2 hereafter referred to as "**S2**" assumes the following set of circumstances:

Total mission velocity increment = 12,599 m/sec.
Trans-Earth injection propulsion stage containment mass is retained for counterweight mass on return leg rotation cycle.

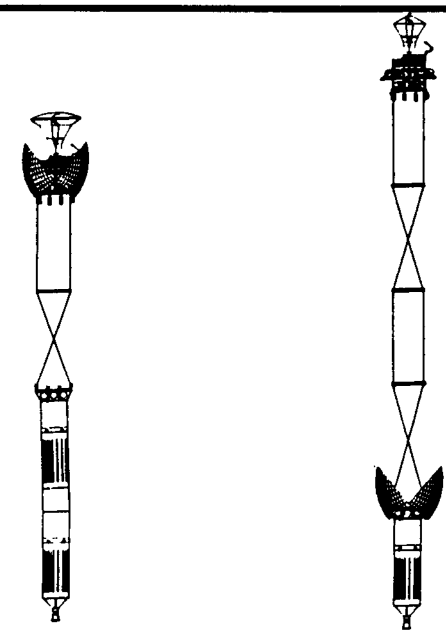
Aerobrake remains attached to MPV and is not transferred as in S1.
Aerobraking maneuver has a propulsive assist = 610 m/sec which requires more fuel than S1.
Trans-Mars injection is one propulsive maneuver.

A **schematic failure analysis** of the tether system indicates that based on the overdesigned tether diameter, two tethers could fail leaving the remaining two tethers to hold the counterweights. In this failure mode the tether system would be retrieved and the mission would continue without artificial gravity.

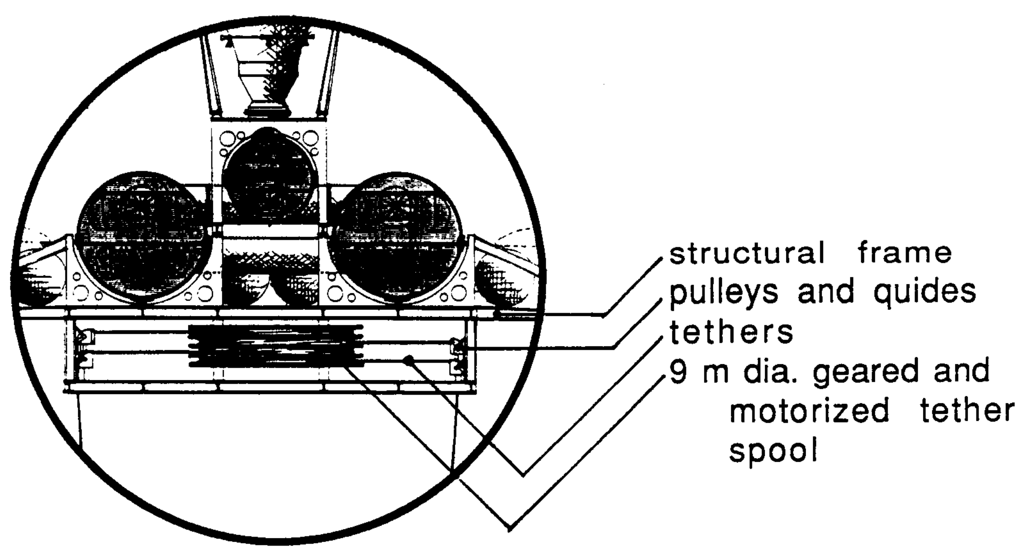
The tether system data contained in this section is the result of cursory research on the subject which included preliminary weight and sizing studies. Further research on this type of system is suggested as an area for future study.

Tether system total weight budget = 13,000 kg.

S1 represents the best case for the tether system based on overall tether length.



	Trans-Mars Coast (S1 Pictured)	Trans-Earth Coast (S1 Pictured)
S1 tether length	73 m	190 m
S2 tether length	95 m	320 m



Tether System Detail

Figure 4-17 MPV Four-Tether System Showing S1
All tether length calculations are based on the counterweight mass at start of rotation cycle, and a minimum radius arm of 107 meters (Baracat 1986) (see figure 4-16).

4.6.6 Propulsion System

The staged propulsion system is designed to accelerate the MPV through the required velocity changes throughout the mission while eliminating the fuel containment mass which is no longer needed. See Appendix C for propulsion system worksheets.

Chemical propulsion data including Isp values and payload to weight ratios are assumed to be consistent with Saturn V technology (Root 1966) and therefore is conservative in nature.

Reaction Control Systems (RCS) are located along the length of the propulsion system which aid in attitude management and spin maintenance.

Operative propellants are liquid hydrogen and liquid oxygen.

Multiple stages per mission phase has been recognized as a posetive design option which may be more closely analyzed in a future design study. In analyzing the propulsion staging of this long and complex misssion it was recognized that there was a significant economy of mass in staging off as much propulsion containment mass as possible for each kg of propellant used. The MME conceptual study of the propulsion system proposes the use of a separate propulsion stage for each major propulsive maneuver, e.g. trans-Mars injection, etc. S1 further proposes two stages within the trans-Mars injection phase. A substantial mass savings was shown to be possible using this strategy.

Propulsion system assessments were wade based on concept design scenarios presented in section 4.6.5.

Technical assistance was provided by Mr. Dennis Fielder for this section of the report. The values in the following figure are taken from spreadsheet programs (Fielder 1988) designed to conceptually analyze propulsion systems. See Appendix C Mars Propulsion System Assessment for S1 & S2.

A schematic failure analysis of the propulsion system indicates that if the propulsion system fails at any time after the trans-Mars injection phase has begun there is little chance for the crew to safely return to Earth. Even assuming a free return flyby, the outbound and inbound travel time could number in the years. A MME with the requirement of supplying contingency consumables and life support for this length of time would be very impractical.

Observations:

S1 represents the best case for the propulsion system based on less overall delivery mass to LEO.

It is interesting to note that without the use of aerobraking at Earth, assuming an all propulsive braking manever, the overall LEO delivery mass would be approximately 10.09 million kg vs. 3.7 million kg for the design presented in this paper.

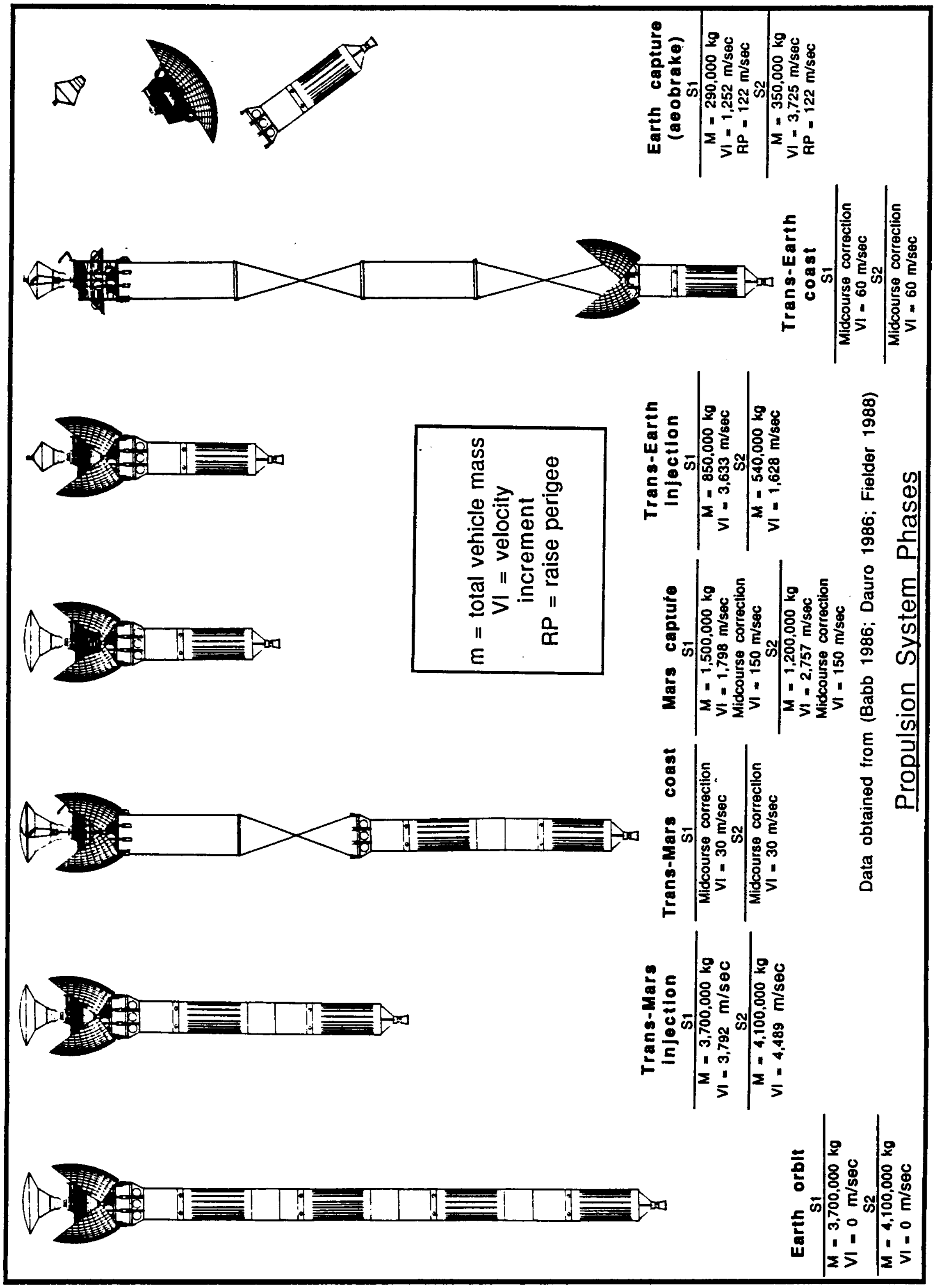


Figure 4-18 MPV Propulsion System Stages

4.6.7 Mass Summary

The following mass summary consists of a general budget set for each subsystem. These figures are not based on detailed mass calculations.

Communication Satellite	3,500 kg
Crew Command Vehicle	50,000
Power System	20,700
Pressurized Environment System	98,000
Structural System	32,000
Folding Aerobrake System	70,000
Tether System	13,000
Subtotal	**287,200**
Propulsion System	
S1	3,412,800
S2	3,812,800
Total	
S1	**3,700,000 kg**
S2	**4,100,000 kg**

5.0 Study Benefits/Areas for Further Development

5.1 Benefits of the MME Project

What was learned?

A substantial amount of research was completed during the MME project which led to the conceptualization of many positive design options.

Reliable/reusable CCV
Four-tether system
Folding aerobrake
Pressurized environment system

5.2 Areas for further development

During the course of this study the following topics were recognized as very significant to future planetary missions which lacked a commensurate body of knowledge:

Life sciences studies to assess the need for partial/full gravity on long duration missions.

Tether technology for large scale applications.

Aerobraking technology for large scale applications.

Nuclear electric power/propulsion for interplanetary uses.

6.0 REFERENCES

Abell, G. O. 1982. *Exploration of the universe.* Philadelphia: CBS College Publishing, pp. 694-95.

Av. Wk. & Space Tech. 1985. Soviets planning 1988 mission to study Martian Phobos. Oct. 1985, pp. 66-67.

Av. Wk. & Space Tech. 1988. NASA to evaluate two suit designs for space station. 11 Jan. 1988, pp. 36-39.

Babb, G. R., and Stump, W. R. 1986. Comparison of mission design options for manned Mars missions. *Manned Mars missions: working group papers.* NASA M002. Vol. 1, pp. 37-52.

Baracat, W.,and Butner, L. 1986. *Tethers in space handbook.* NASA office of spaceflight advanced programs office, Code MT, NASA-HQ. NASW 3921.

Boston, P. J., ed. 1984. *The case for Mars.* Science and Technology Series, vol. 57. San Diego: American Astronautical Society.

Carr, G. 1988. S.I.C.S.A., Houston. Series of program and design reviews. Fall 1987-Spring 1988.

Carr, M. H. 1981. *The surface of mars.* New Haven: Yale Univ. Press.

------ 1986. Science objectives. *Manned Mars missions: working group papers.* NASA M002, vol. 1, pp. 3-12.

Clifford, S. 1985. Workshop on water on Mars. LPI tech. report 85-03. Houston: LPI.

Coomes, E. P., et al. 1986. Pegasus: a multi-megawatt nuclear propulsion system. In *Manned Mars missions: working group papers,* NASA M002, vol. 2, pp.769-85.

Dauro, V. A. 1986. Aerobraking. In *Manned Mars missions: working group papers,* NASA M002. vol. 1, pp. 21-36.

Du Pont Co. 1976. Characteristics and uses of Kevlar 29 aramid. A preliminary information memo, no. 375, September.28, 1976. Wilmington, Delaware.

------ 1983. A guide to cutting and machining materials containing Kevlar aramid fiber. A preliminary information memo, no. 446, December, 1983. Wilmington, Delaware.

Eagle Engineering 1986. An Aggressive Future Space Program. NASA Contract No. NAS9-17317, Eagle Engineering Report No. 86-118. January 24, 1986. Houston, TX.

Fielder, D. E. 1988. Mars propulsion system assessment. S.I.C.S.A., Houston, Texas.
(A spreadsheet program run on Lotus 1-2-3, release 1A. Calculates stage mass given payload budget and velocity increments for a staged propulsion system.)

------. 1988. Rocket equation. S.I.C.S.A., Houston, Texas.
(A program, written in "basic" language, which implements the basic rocket equation (ΔV_i.) The program is optimised for single stage vehicles.)

Fuller, S. 1986. Mars landing mission: a structural approach. In *Manned Mars missions: working group papers,* NASA M002, vol. 2, pp. 733-44.

Galecki, D. L., and Patterson, M. J. 1987. *Nuclear powered Mars cargo transport mission utilizing advanced ion propulsion.* NASA Technical Memorandum 100109.

Garrison, P. W.; Nock, K. T.; and Jones, R. 1984. Nuclear electric propulsion (nep) spacecraft configuration study. In *Orbit-raising and maneuvering propulsion: research status and needs,* ed. L. H. Caveny, pp. 544-56. New York: AIAA.

General Dynamics 1986. OTV Concept Definition and Systems Analysis Study. NASA Contract No. CR DGSS-SP-86-011.

Glass, B. P. 1982. *Introduction to planetary geology.* Cambridge, Mass.: Cambridge Univ. Press.

Goldman, N. C. 1985. The Mars base: international cooperation. In *The case for Mars 2,* ed. C. P. McKay,.pp. 65-71. San Diego: American Astronautical Society.

Grandjean, E. 1987. *Fitting the task to the man.* Philadelphia: Taylor & Francis.

Harvey, B. 1986. Phobos lander Mars project. *Spaceflight.* 28:113-114.

Hoffman, S. J. 1986. A review of manned mars missions in the space station era. Paper presented at AIAA Space Station in the 21st Century Conference, 4 Sep. 1986, at Reno Nevada.

Kaufman, H. R., and Robinson, R. S. 1984. Electric thruster performance for orbit-raising and maneuvering. In *Orbit-raising and maneuvering propulsion: research status and needs,* ed. L. H. Caveny, pp. 303-26. New York: AIAA.

Letaw, J. R., and Clearwater, S. 1986. *Radiation shielding requirements on long-duration space missions.* Severn Communications Corporation for the Naval Research Laboratory, Washington, D.C. Contract #N00014-85-C-2200.

Martin Marietta. OTV Contract Extension Final Review. NASA Contract No. NAS8-36108

McKay, C. P., ed. 1985. *The case for Mars* 2. Science and Technology Series, vol. 62. San Diego: American Astronautical Society.

Mutch T.A. 1976. *The geology of mars.* Princeton: Princeton Univ. Press.

NASA. 1987. *Man-system integration standards.* NASA-STD-3000, vol. 1, sec. 5.

National Commission on Space 1986. *Pioneering the space frontier.* New York: Bantam Books.

Neihoff, J. 1985. Manned mars mission design. Paper presented at Steps to Mars, a joint AIAA/Planetary Society Conference, 16 Jul. 1985, at National Academy of Sciences, Washington, D.C.

Nock, K. T., and Friedlander, A. L. 1986. Elements of a mars transportation system. Paper presented at the 37th Congress of the International Astronautical Federation, 4-11 Oct. 1986, at Innsbruck, Austria.

O'Leary, B. 1985. Phobos and Deimos: concept for early human mission for resources and science. Paper presented at the 7th Princeton/AIAA/SSI Conference on Space Manufacturing.

------. 1987. Mars 1999: A concept for high leverage water and fuel processing on Phobos and Deimos. Paper presented at the May 1987 Princeton/AIAA/SSI Conference on Space Manufacturing.

Page, M. 1986 Earth-to-orbit launch vehicles for manned Mars mission applications. In *Manned Mars missions: working group papers*, NASA M002, vol. 1, pp. 228-38.

Phillips, P. G. Nuclear electric propulsion freighter: a point design. Paper presented at The Case for Mars 3 Conference, 18-22 Jul., 1987, at University of Colorado, Boulder, Colorado.

Planetary Society. 1984. Manned lunar, asteriod, and mars missions. Paper prepared by Science Applications International Corp., Sep. 1984. Schaumburg, Illinois: SAIC.

Ride, S. K. 1987. *Leadership and America's future in space.* A report to the NASA Administrator, Aug. 1987.

Root, M. W. 1965. Application of Saturn S-IVB Apollo systems to planetary exploration. In *Post Apollo space exploration*, ed. F. Narin. Advances in the Astronautical Sciences, vol. 20, part 1, pp. 207-43. Tarzana, California: AAS Publications Office.

Rose, R. G. 1987. Rockwell shuttle operations company. Meeting notes. 9 Mar. 1987.

Rudey, R., et al. 1987. *The evolution of space station power systems technology.* Sundstrand Corp. Rockford, IL. IAF 87-226.

Science. 1978. The composition of phobos: evidence for carbonaceous chondritic surface from spectral analysis. 199:64-66.

Singer, S. F. 1984. The Ph-D proposal: a manned mission to Phobos and Deimos. In *The Case For Mars*, ed. P. J. Boston, pp.39-65. San Diego: American Astronautical Society.

------ 1986. Phobos and deimos: A base for sampling the martian past. In *MECA symposium on Mars: evolution of its climate and atmosphere,* p. 98. Houston: Lunar and Planetary Institute.

Sprengle, V. 1987. Comparison between photovoltaic and solar dynamic power system concepts for low earth orbit applications. Paper presented to the 38th Congress of the International Astronautical Federation, 10-17 Oct. 1987, at Brighton, United Kingdom.

Stump, W. R.; Babb, G. R.; and Davis, H. P. 1986. Mars lander survey. In *Manned Mars missions: working group papers.* , NASA M002, vol. 1, p. 239-51.

Toulmin, P., et al. 1977. Composition of martian samples from Viking 1 & 2 lander sites. *Journal of Geophysical Research.* 82:4625.

Welch, S. M., and Stoker, C. R. 1985. Concept development for a Mars research station. In *The case for Mars*, ed. P. J. Boston San Diego: American Astronautical Society.

Young, A. C. 1986a. Mars mission concepts and opportunities. In *Manned Mars missions: working group papers*, NASA M002, vol. 1, pp. 103-13.

------ 1986b. Mission and vehicle sizing. In *Manned Mars missions: working group papers*, NASA M002, vol. 1, pp. 87-102.

APPENDIX A ICTV PERFORMANCE DATA

I C T V   M A S S   S U M M A R Y

Power Subsystem (3000 kWe)		23,000 kg	
Reactor/Power Conversion	5000 kg		
Shielding	3000 kg		
Radiator	15,000 kg		
Propulsion (Electric and Chemical)		19,600 kg	
Thrusters & Power Processor	12,800 kg		
Tankage	5800 kg		
Chemical RCS	1000 kg		
Structural		2100 kg	
Guidance and Navigation		500 kg	
Communications		100 kg	
Data Management		100 kg	
Total: Dry Mass			45,400 kg
RCS Propellants (Hydrazine)	3000 kg		
Xenon Propellants	111,250 kg		
Payload	153,000 kg		
TOTAL: LEO Mass			312,650 kg

P E R F O R M A N C E   S U M M A R Y

Reactor size (kWe): 3000
Power Processor Efficiency: 90%
Thruster Efficiency: 85%
Overall Efficiency: 76.5%

Payload mass: 153,000 kg
Propellant mass: 114,250 kg
Vehicle mass: 45,400 kg

Thrusters: 20 50-cm. ion
Propellant: Xenon
Isp: 5000 sec.
Thrust: 70.2 N

Trajectory: NSO slow spiral escape
Trip time: approx. 600 days LEO to LMO
Return trip: approx. 300 days LMO to LEO

Note: These are approximate figures only and can vary according to the assigned mission parameters.

ICTV performance data
(Coomes et al. 1986; Fielder 1988; Galecki and Patterson 1987; Garrison Nock and Jones 1984; Kaufman and Robinson 1984; Nock and Friedlander 1986; Phillips 1987)

APPENDIX B CCV WEIGHT SUMMARY

CREW COMMAND VEHICLE WEIGHT SUMMARY (in kg)

Mars Vicinity: Aerobrake remains on surface of Mars, Ascent stage has lightweight "Landing Gear" for Phobos/Deimos docking.

Earth Return: Crew of six, return of Crew Command Vehicle to Low Earth Orbit.

ASCENT Manned Areas	Mars Vicinity		Earth Return
Primary Structure	380		380
Couches, Restraints	55		55
Hatches/Windows	70		70
Docking Provisions	75		75
Panels, Supports	20		20
Elec Power System	435		435
(2kW fcell)			0
EPS Distribution	105		105
GN&C	200		200
Instrumentation	85		85
Life Support System	600		600
4 Crew	320	6 Crew	480
Manned Capsule Total	2,345		2,505
Lander/Launcher System			
Propulsion System (Isp 460)	2,300		2,300
Aerobrake	0		0
Landing Gear	135		135
RCS-- Dry	215		215
RCS-- Propellant	135		135
Primary Structure	450		450
Science Payload	575		575
Lander/Launcher Total	3,810		3,810
> Total Stage Dry Weight	6,155		6,315

Propellant Requirements	Mars Sfc to PO		TMI to EOI	
Delta V (m/s)	5,075		1,725	
Total Prop Required		12,785		2,940
Oxygen to Hydrogen Ratio	7:1		7:1	
Total Oxygen Weight	10,959		2,520	
Total Hydrogen Weight	1,826		420	
Cubic Meters of Oxygen	9.6		2.2	
Cubic Meters of Hydrogen	26.5		6.1	
> Total Vehicle Stage Weight		18,940		9,255

DESCENT

Manned Areas		
Life Support Consumables		135
Science Packages, Incl Rover		1,000
Subtotal		1,135
Lander/Launcher System		
Propulsion System (Isp 460)		2,300
Landing Gear		320
Aerobrake		6,000
Primary Structure		900
Subtotal		9,520
> Descent Payload Weight (Including Ascent Wt)		29,595
Propellant Requirements	**Phobos Orbit to MS**	
Delta V (m/s)	2,400	
Total Ascent Prop Required		20,775
Oxygen to Hydrogen Ratio	7:1	
Total Oxygen Weight	17,807	
Total Hydrogen Weight	2,968	
Cubic Meters of Oxygen	15.7	
Cubic Meters of Hydrogen	43.1	
> TOTAL VEHICLE WT @ DESCENT		50,370

(Eagle Engineering; Fielder 1988; Stump et al. 1986)

ORIGINAL PAGE IS
OF POOR QUALITY

ORIGINAL PAGE IS
OF POOR QUALITY

APPENDIX C RADIATION DESIGN REQ'S

NASA-STD-3000

**IONIZING RADIATION
DESIGN REQUIREMENTS**

Constraints in REM	Depth (5 cm)	Eye (0.3 cm)	Skin (0.1 mm)
30 days	25 Rem**	100 Rem	150 Rem
Annual	50 Rem	200 Rem	300 Rem
Career	100-400* Rem	400 Rem	600 Rem

Reference: 206, Table 7-3

- The career depth dose-equivalent limit is based upon a maximum 3% lifetime risk of career mortality. The total dose-equivalent yielding this risk depends on sex and age at start of exposure. The career dose-equivalent limit is approximately equal to:
  200 + 7.5 (age minus 30) rem, for males, up to 400 rem maximum
  200 + 7.5 (age minus 30) rem, for females, up to 400 rem maximum

** Rem = radiation absorbed dose, in rads, multiplied by a quality factor, Q, to account for the relative biological effectiveness (RBE) of different types of radiation. For planning purposes, Q = 1.2

Note: The testes have been removed as a critical organ (compare to Figure 5.7.2.2.1-1). This is based on indications from NCRP Committee 75 that despite concern for the radiation dose equivalent absorbed by the testes data does not warrant its consideration as a separate critical organ.

MSIS-177

Figure 5.7.2.2.1-2. Ionizing Radiation Exposure Limits From Space Flight Being Proposed by National Council on Radiation Protection Committee 75 (1986)

a. Radiation Protection - The design of the space module shall include the necessary radiation protection features (shielding, radiation monitoring and dosimetry, "storm shelter", etc.) for all expected missions to ensure that the crew dose rates are kept as low as reasonably achievable (*ALARA* levels) and that the maximum allowable dose limits are not exceeded.

b. Protection Consistent With Orbit - Radiation protection provisions shall be consistent with the flight path, altitude, and inclination of the orbit.

c. Use of Onboard Mass - The design and layout of the space module shall make optimal use of onboard mass as radiation shielding, especially for missions where dose rates are expected to be appreciable.

d. Solar Radiation Warning - An alert system for particle events associated with solar flares shall be provided for missions where SPEs pose a threat (planetary, polar, or geosynchronous orbits).

e. Radiation Contingency Plans - Contingency plans for crew protection during solar particle events and other emergencies shall be provided for missions where SPEs pose a threat.

f. Mission Radiation Control Program - A mission radiation control program shall be instituted to establish radiation exposure limits, procedures, and responsibilities consistent with the expected mission environment and duration of orbital stay in order to keep radiation exposures to crew at ALARA levels.

g. Radioactive Waste Disposal - Safe procedures shall be established for the handling and disposal of radioactive waste or radioactively contaminated materials.

h. Cumulative Crewmember Radiation Dose - The radiation dose equivalent accumulated by each spacecraft crewmember shall be monitored throughout the active career of all crewmembers. Thus career, as well as mission dose equivalent levels shall be kept ALARA, thereby ensuring that the maximum career dose equivalent limit shall not be exceeded.

(NASA 1987)

ORIGINAL PAGE IS
OF POOR QUALITY

APPENDIX D PROPULSION SYSTEM ASSES'T

File:OPT1
MARS PROPULSION SYSTEMS ASSESSMENT
Version xx- December 1, 1987

Basic Rocket Eqn.: V=I.g.Log e (Wo/Wbo) or V=I.g.Log e (Wo/Wo-Wp)
MARS2C
Mission Sequence - EO, TMI, MOI, MO, TEI, EOI, EO (plus Midcourse corrections)
(Mcc and MOI combined as one stage/9145 fps and MO, TEI and Mcc combined

Item #	Mission Sequence (a)	Net Return Payload (b)	Sequence Start Mass (c)	Increment Mass Change (d)	Sequence Velocity Increment (e)	Fuel Expended (f)	Fuel Containment Mf=.9 (g)	Stage Mass Mf=.9 (h)
1.00	EO	8194533.82	8194534					
2.00	TMI	8194533.82	8194534		6220.00	2829761.5	314417.9	3144179.4
3.00	Mcc	5364772.35	5050354		6220.00	1744003.8	193778.2	1937782.0
4.00	MOI	3306350.59	3112572		5899.00	1029804.2	114422.7	1144226.9
5.00	MO	2082768.16	1968345	83500.00	0.00	0.0	0.0	0.0
6.00	TEI	1884845.47	1884845		11919.00	1047817.8	116424.2	1164242.0
7.00	Mcc	837027.63	720603	52356.50	0.00	0.0	0.0	0.0
8.00	CcvD	668246.93	668247	19380.00	0.00	0.0	0.0	0.0
9.00	EO_A/B	648866.93	648867	0.00	0.00	0.0	0.0	0.0
10.00	EOI	648866.93	648867		0.00	0.0	0.0	0.0
11.00	EOC	648866.93	648867		600.00	25980.2	2886.7	28866.9
12.00	EO	622886.69	620000		0.00	0.0	0.0	

Note: TEI Stage carried through MO aand TEI for Counter balance mass.
Notes:

C-9 = Payload mass to aerobrake to earth orbit (600 X 600)
D-8 = CCV Mass to aerobrake to earth orbit (Deployed after TEI/Mcc)
D-5 = Mass left in Mars vicinity by CCV
E-9 = 4107 ft/sec Aerobrake to Earth Orbit (600 x 600)
H-11 = Propulsion system mass embodied in Sc envelope during EOI A/B
E-6 = MO + TEI + Mcc (500 + 5341 + 200 = 6041 fps)

Propulsion system assessment for concept design scenario 1 (Babb 1986; Dauro 1986; Fielder 1988)

ORIGINAL PAGE IS
OF POOR QUALITY

File:OPT2
MARS PROPULSION SYSTEMS ASSESSMENT (ft/lb/sec)
Version xx- December 1, 1987

Basic Rocket Eqn.: V=I.g.Log e (Wo/Wbo) or V=I.g.Log e (Wo/Wo-Wp)

Mission Sequence - EO, TMI, MOI, MO, TEI, EOI, EO (plus Midcourse corrections)

Item #	Mission Sequence (a)	Net Return Payload (b)	Sequence Start Mass (c)	Increment Mass Change (d)	Sequence Velocity Increment (e)	Fuel Expended (f)	Fuel Containment Mf=.9 (g)	Stage Mass Mf=.9 (h)
1	EO	9125682.195	9125682					
2	TMI	9125682.195	9125682		14727	5765544.0	640616.0	6406159.9
3	Mcc	3360138.242	2719522		100	18458.4	2050.9	20509.3
4	MOI	2701063.855	2699013		9045	1241289.4	137921.0	1379210.4
5	MO	1457723.523	1319802	83500	500	41390.5	4598.9	45989.4
6	TEI	1194911.985	1190313		5341	362967.4	40329.7	403297.1
7	Mcc	827345.6510	787016		200	10647.3	1183.0	11830.3
8	EO-A/B	776368.6676	775186	19380	2000	96244.8	10693.9	106938.7
9	EOI	659560.8000	648867		0	0.0	0.0	0.0
10	EOC	648866.9292	648867		600	25980.2	2886.7	28866.9
11	EO	622886.6929	620000					
12								

Notes:

e-8 = 12,221 ft/sec total, 2,000 ft/sec propulsive
d-5 = assumed mass deployed/expended in Mars orbit

f-10 = velocity budget to circularize EO to 800 from 400 to 1600 km

Propulsion system assessment for concept design scenario 2 (Babb 1986; Dauro 1986; Fielder 1988)

File:MARSALLPROP

MARS PROPULSION SYSTEMS ASSESSMENT (ft/lb/sec) e 2.72 Mf 0.9

Version xx- December 1, 1987

Basic Rocket Eqn.: V=I.g.Log e (Wo/Wbo) or V=I.g.Log e (Wo/Wo-Wp)

Mission Sequence - EO, TMI, MOI, MO, TEI, EOI, EO (plus Midcourse corrections)

Item #	Mission Sequence (a)	Net Return Payload (b)	Sequence Start Mass (c)	Increment Mass Change (d)	Sequence Velocity Increment (e)	Fuel Expended (f)	Fuel Containment Mf=.9 (g)	Stage Mass Mf=.9 (h)
1	EO	20011970.12	20011970					
2	TMI	20011970.12	20011970		14727	12643426.6	1404825.2	14048251.8
3	Mcc	7368543.489	5963718		100	40477.9	4497.5	44975.5
4	MOI	5923240.370	5918743		9045	2722059.1	302451.0	3024510.1
5	MO	3196683.708	2894233	83500	500	94101.3	10455.7	104556.9
6	TEI	2716631.442	2706176		5341	825206.1	91689.6	916895.6
7	Mcc	1880969.674	1789280		200	24206.6	2689.6	26896.2
8	EO-A/B	1765073.544	1762384	19380	12221	984723.3	109413.7	1094137.0
9	EOI	758280.6289	648867		0	0.0	0.0	0.0
10	EOC	648866.9292	648867		600	25980.2	2886.7	28866.9
11	EO	622886.6929	620000					
12								

Notes:

e-8 = 12,221 ft/sec all propulsive maneuver

d-5 = assumed mass deployed/expended in Mars orbit

f-10 = velocity budget to circularize EO to 800 from 400 to 1600 km

Propulsion system assessment for all propulsive earth orbit capture (Babb 1986; Dauro 1986; Fielder 1988)

ORIGINAL PAGE IS
OF POOR QUALITY

APPENDIX E PLANETARY INFORMATION

PLANET	DIAMETER km	DIAMETER Earth = 1	MASS (Earth = 1)	MEAN DENSITY (g/cm^3)	ROTATION PERIOD (days)	INCLINATION OF EQUATOR TO ORBIT	OBLATENESS	SURFACE GRAVITY (Earth = 1)	ALBEDO	VISUAL MAGNITUDE AT MAXIMUM LIGHT*	VELOCITY OF ESCAPE (km/s)
Mercury	4878	0.38	0.055	5.43	58.6	0°0	0	0.38	0.106	−1.9	4.3
Venus	12,104	0.95	0.82	5.24	−243.0	177.4	0	0.91	0.65	−4.4	10.4
Earth	12,756	1.00	1.00	5.52	0.9973	23.4	1/298.2	1.00	0.37	—	11.2
Mars	6794	0.53	0.107	3.9	1.026	25.2	1/164	0.38	0.15	−2.0	5.0
Jupiter	142,796	11.2	317.8	1.3	0.41	3.1	1/16	2.53	0.52	−2.7	60
Saturn	120,000	9.41	94.3	0.7	0.43	26.7	1/9.2	1.07	0.47	+0.7	36
Uranus	50,800	3.98	14.6	1.3	−0.65	97.9	1/30	0.92	0.50	+5.5	21
Neptune	50,450	3.81	17.2	1.5	0.77	29	1/40	1.18	0.5	+7.8	24
Pluto	3400	0.27	0.0023	0.5 (?)	6.387	90	?	0.03	0.5	+15.1	1

*At mean opposition for superior planets.
Adapted from *The Astronomical Almanac* (U. S. Naval Observatory), 1981.

(Abell 1982)

ORIGINAL PAGE IS OF POOR QUALITY

PLANET	SATELLITE	DISCOVERED BY	MEAN DISTANCE FROM PLANET (km)	SIDEREAL PERIOD (Days)	ORBITAL ECCEN-TRICITY	DIAMETER OF SATELLITE* (km)	MASS† (Planet = 1)	APPROXIMATE MAGNITUDE AT OPPOSITION
Earth	Moon	—	384,404	27.322	0.055	3476	0.0123	−12.5
Mars	Phobos	A. Hall (1877)	9,380	0.319	0.021	25	(2.7×10^{-8})	+12
	Diemos	A. Hall (1877)	23,500	1.262	0.003	13	4.8×10^{-9}	13
Jupiter‡	XIV	Voyager II (1979)	129,000	0.297	0	<40	(10^{-10})	18–19
	V Almalthea	Barnard (1892)	181,300	0.498	0.003	240	(2×10^{-9})	13
	I Io	Galileo (1610)	421,600	1.769	0.000	3640	4.7×10^{-5}	5
	II Europa	Galileo (1610)	670,900	3.551	0.000	3130	2.5×10^{-5}	6
	III Ganymede	Galileo (1610)	1,070,000	7.155	0.002	5270	7.8×10^{-5}	5
	IV Callisto	Galileo (1610)	1,880,000	16.689	0.008	4840	5.6×10^{-5}	6
	VI Himalia	Perrine (1904)	11,470,000	250.57	0.158	(170)	(8×10^{-10})	14
	VII Elara	Perrine (1905)	11,800,000	259.65	0.207	(40)	(4×10^{-11})	18
	X Lysithea	Nicholson (1938)	11,850,000	263.55	0.130	(10)	(1×10^{-12})	19
	XIII Leda	Kowal (1974)	11,110,000	239.2	0.147	(8)	(5×10^{-14})	20
	XII Ananke	Nicholson (1951)	21,200,000	631.1	0.169	(10)	(7×10^{-13})	18
	XI Carme	Nicholson (1938)	22,600,000	692.5	0.207	(15)	(2×10^{-12})	19
	VIII Pasiphae	Melotte (1908)	23,500,000	738.9	0.378	(25)	(8×10^{-12})	17
	IX Sinope	Nicholson (1914)	23,700,000	758	0.275	(15)	(2×10^{-12})	18
Saturn§	Mimas	W. Herschel (1789)	185,500	0.942	0.020	390	6.6×10^{-8}	13
	Enceladus	W. Herschel (1789)	237,900	1.370	0.004	500	1.3×10^{-7}	12
	Tethys	Cassini (1684)	294,700	1.888	0.000	1050	1.1×10^{-6}	10
	Dione	Cassini (1684)	377,400	2.737	0.002	1120	1.8×10^{-6}	10
	Rhea	Cassini (1672)	526,700	4.518	0.001	1530	4×10^{-6}	10
	Titan	Huygens (1655)	1,222,000	15.945	0.029	5120	2.3×10^{-4}	8
	Hyperion	Bond (1848)	1,481,000	21.277	0.104	310	(2×10^{-7})	14
	Iapetus	Cassini (1671)	3,560,000	79.331	0.028	1440	3.3×10^{-6}	11
	Phoebe	W. Pickering (1898)	12,930,000	550.45	0.163	40	(7×10^{-10})	16
Uranus	Miranda	Kuiper (1948)	123,000	1.414	0	(200)	1×10^{-6}	17
	Ariel	Lassell (1851)	191,700	2.520	0.003	(600)	1.5×10^{-5}	14
	Umbriel	Lassell (1851)	267,000	4.144	0.004	(400)	6×10^{-6}	15
	Titania	W. Herschel (1787)	438,000	8.706	0.002	(1000)	5×10^{-5}	14
	Oberon	W. Herschel (1787)	585,960	13.463	0.001	(900)	3×10^{-5}	14
Neptune	Triton	Lassell (1846)	353,400	5.877	0.000	6000	3×10^{-3}	13
	Nereid	Kuiper (1949)	5,560,000	359.881	0.749	(500)	(10^{-6})	19
Pluto	Charon	Christy (1978)	17,000	6.387	0	(1200)	(0.1)	17

*A diameter of a satellite given in parentheses is estimated from the amount of sunlight it reflects.
†A mass of a satellite given in parentheses is estimated from its size and an assumed density.
‡Does not include two satellites discovered by Voyager; see Chapter 18.
§Does not include at least seven satellites discovered by Voyager; see Table 18.2.
Orbital data from *The Astronomical Almanac* (U. S. Naval Observatory).
Other data compiled from various sources.

Satellites of the planets (Abell 1982)

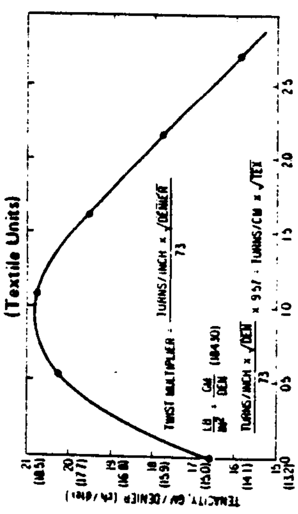
APPENDIX F TETHER DESIGN DATA

COMPARATIVE YARN PROPERTIES

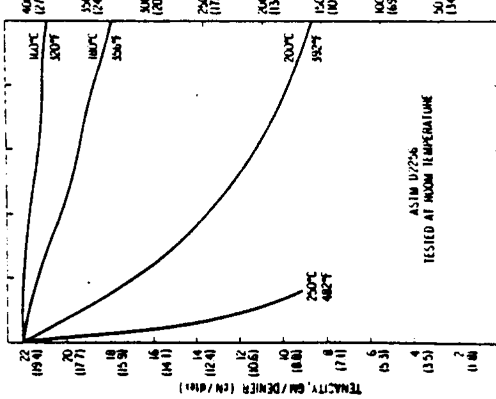
	KEVLAR® 29 Aramid	DU PONT Nylon Type 728	DACRON® Polyester Type 68	"E-HTS" Glass	Stainless Steel
Tensile Strength, lb/in^2	400,000**	143,000**	162,500**	350,000***	250,000
(MPa)*	(2758)	(985)	(1120)	(2412)	(1724)
Modulus, lb/in^2	9,000,000	800,000	2,000,000	10,000,000	29,000,000
(MPa)	(62000)	(5512)	(13780)	(68900)	(199800)
Elongation to Break, %	4.0	18.3	14.5	3.5	2.0
Density, lb/in^3	0.052	0.041	0.050	0.092	0.284
(g/cm^3)†	(1.44)	(1.14)	(1.38)	(2.55)	(7.83)

*MPa = MN/m^2 = lb/in^2 x 6.895×10^{-3}
**Unimpregnated twisted yarn test -- ASTM D2256
***Impregnated strand test -- ASTM D2343
†g/cm^3 = lb/in^3 x 27.68

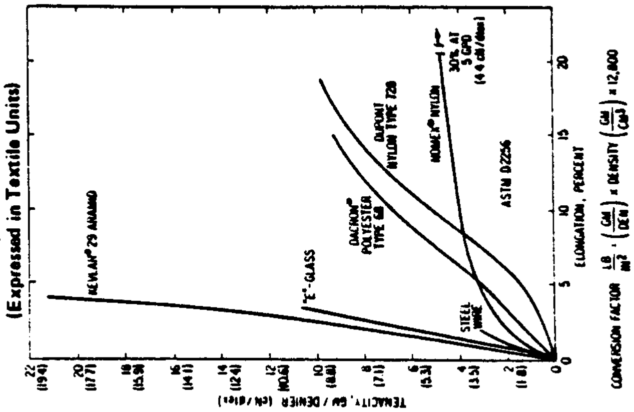
THE EFFECT OF TWIST ON THE TENSILE STRENGTH OF KEVLAR® 29 ARAMID
(Textile Units)



THE EFFECT OF TEMPERATURE ON THE TENSILE STRENGTH OF KEVLAR® 29 ARAMID



STRESS-STRAIN BEHAVIOR OF YARNS
(Expressed in Textile Units)



Tether design data (DuPont 1976)